CHAPTER 2. PART 27
AIRWORTHINESS STANDARDS
NORMAL CATEGORY ROTORCRAFT

SUBPART E - POWERPLANT

POWERPLANT - GENERAL

AC 27.901. § 27.901 (through Amendment 27-20) INSTALLATION.

a. Section 27.901(a)

(1) Explanation. Paragraph (a) provides a definition of parts of rotorcraft for which safety requirements are set forth under the general title, SUBPART E - POWERPLANT. These parts include not only major propulsion elements and power transmission components but also controls, instruments, safety devices, including fire protection and other devices to protect personnel, and critical flight structure in event of fires.

(2) Procedures. To ensure that no certification aspect is overlooked in establishing compliance, certification engineers should make at least an informal breakdown of all components of the rotorcraft, assigning responsibility to powerplant certification engineers of all items within the above definition. While this procedure is usually straightforward, the following items of FAA/AUTHORITY powerplant responsibility are listed to minimize questions regarding authority and responsibility.

(i) Drive system components. All parts of the transmission, clutches, shafting, including the driveshafts (masts) of main and auxiliary rotors, powerplant cooling components, and powerplant instrumentation requirements under §§ 27.1305, 27.1337, 27.1543, 27.1549, 27.1551, 27.1553, 27.1555, and 27.1583.

NOTE: The division of responsibility between FAA/AUTHORITY airframe engineers and FAA/AUTHORITY powerplant engineers (in accordance with FAA/AUTHORITY practice) regarding the driveshaft is at the flange or spline interface between the driveshaft and the rotor hub. Rotor hubs, controls, blades, and associated components are the airframe engineers' responsibility. (Industry practice may not agree with this concept.)

(ii) Engines, except for mount structure.

(iii) Auxiliary power units, except for mount structure.

(iv) Combustion heaters, except for downstream ventilation air ducting, mixing, and distribution systems and for electrical aspects of controls and safety devices.

(v) Water/alcohol or other fluid power augmentation systems.

(vi) Engine induction systems including induction icing and snow ingestion, and exhaust systems, including exhaust shrouds and drains.

(vii) All fuel systems, including those serving engines, auxiliary power units, combustion heaters, power augmentation systems, etc., and vents and drains for those systems.

(viii) Oil systems for engines, auxiliary power units, rotor drive transmissions, and gearboxes, including grease lubrication.

(ix) Cooling aspects of engines, rotordrive transmissions and gearboxes, and auxiliary power units.

NOTE: Electrical generating equipment and hydraulic component cooling may be the responsibility of the systems and equipment engineer provided agreement is established among responsible personnel.

(x) Rotor brakes, except hydraulic and electrical aspects and structural aspects of nonrotating brake components.

(xi) Fire protection, including firewalls, fire extinguisher systems, fire detector systems, flammable fluid lines, fittings, and shutoff valves. The powerplant engineer has responsibility for evaluating compliance with §§ 27.861 and 27.863 as it pertains to fuel and oil systems.

(xii) Engine and transmission cowling and covering, including latches.

(xiii) Powerplant flexible controls.

(xiv) Powerplant accessories.

(xv) Pneumatic systems (engine bleed air) within the engine compartments, including shut-off valves and engine isolation features of bleed systems.

(xvi) Powerplant aspects of instrument markings and powerplant aspects of flight manuals, including limitations, normal and emergency procedures, engine performance; powerplant aspects of maintenance manuals, with emphasis on the limitations section of the manual and verification of the limitations established under § 27.1521.

b. <u>Section 27.901(b).</u>

(1) Explanation. Paragraph (b) requires that the various powerplant components and systems be investigated for general airworthiness.

(2) Procedures.

(i) Each item of the powerplant area of responsibility should be shown to be suitable for its intended purpose and installed to operate satisfactorily and safely between normal inspections and overhauls. Accessories mounted on engine or transmission drive pads should be determined to be compatible with the pad limits, including fit and speed range, overhang moment loads, running torque and static torque. This latter term pertains to protection of the engine or transmission which drives the accessory from damage to be expected from malfunction of the accessory. This protection is usually supplied by providing a shear section in the accessory drive shaft designed to fail before exceeding the static torque limit of the engine or transmission driving component. Note that when evaluating the strength of the mechanical shear section, material allowables quoted in materials handbooks should not be used since these are minimum strength values. Shear sections should consider maximum strength values to be expected which are on the order of 130 percent of the minimum strength values. Also, it should be verified that design data for shear sections are dimensioned to limit the maximum diameter as well as the minimum diameter. Installation of starter-generators may also require verification that horsepower extraction limits are not exceeded. Special flightcrew instructions in the flight manual to monitor generator load or to disconnect electrically loaded items to protect accessory or engine-transmission pad limits should be avoided.

(ii) Environmental qualification requires consideration or protection against adverse effects of extremes of cold weather, salt and sand/dust atmosphere, altitude effects, etc. Most powerplant components are subjected to many of these aspects during the individual qualification tests; however, satisfactory overall integrated system performance under these adverse conditions must be verified. Cold weather testing should include verification that lubricating oils and greases function properly, and that engine starting procedures are safe and do not impose excessive loads on accessories, engines, or drive system components. Powerplant engineers should coordinate compliance efforts in this area with system engineer's investigations of compliance with §§ 27.1301 and 27.1309. Full-scale rotorcraft operations in cold weather should be required, including at least some exposure in the range of -10° to -20° F if the aircraft is to be certified to these ambients. Cold soak or overnight exposure to cold weather is appropriate followed by starting and pre-takeoff procedures in accordance with the flight manual. Attention should be given to the practicality of important mandatory inspection procedures as affected by cold weather.

(iii) Accessibility for maintenance should be reviewed. Typically, some maintenance activities must involve disassembly or removal of adjacent components. This should be avoided if repetitive activity can jeopardize the performance of critical or safety-related equipment. Verify that easy access exists to items such as oil system sight gauges or dip sticks, filler ports and drain valves for engines, auxiliary propulsion units, transmissions, fuel tanks and filters, etc.

(iv) Electrical interconnections to prevent difference of potential should be provided in the form of grounding straps or wires sized to carry the currents to be expected. Verify that the attachments for these grounding devices are not compromised by paint or zinc chromate which will tend to electrically insulate the engine or component. Note that engine mount structure should not be accepted as a grounding device since electrical current will cause corrosion at attach points.

(v) Axial and radial expansion of turbine engines is usually not a problem unless redundant mount arrangements are used. Special expansion provisions are usually required if engine components other than mounting points are attached to bulkheads, firewalls, other engines, or drive system components. Engine output shaft axial or bending loads due to thermal expansion and to deflection of supports under ground or flight loads should be checked. Other components of concern are compressor inlet flanges, exhaust ducts, and rigid fluid or air lines between aircraft structure and the engine. The engine installation data will provide limit loads to be considered for parts of the engine which normally are attached to airframe components.

c. Section 27.901(c).

(1) Explanation. Paragraph (c), in conjunction with the installation manual requirements of § 33.5, is intended to assure compliance with the detail installation requirements developed by the engine manufacturer to assure safe, continued operation of the engine.

(2) Procedure. Compliance with most of the detail requirements in the engine installation manual can be established by test or by design features and arrangements negotiated between the rotorcraft manufacturer and the FAA/AUTHORITY powerplant engineer. Some aspects, usually involving inlet and/or exhaust distortion limitations, vibration limitations and aircraft/engine interface items may require direct assistance and information from the engine manufacturer to determine that compliance with the installation manual exists. Fuel control/engine/rotor system torsional matching is usually a developmental problem to be worked out before presentation of the rotorcraft to the FAA/AUTHORITY; however, final flight tests for surge or stall, torsional stability, and acceleration/ deceleration schedules may require direct coordination among FAA/AUTHORITY installation engineers, engine manufacturers' representatives, and the FAA/AUTHORITY engine certification engineers. Reciprocating, carburetor equipped engines usually require a particular carburetor configuration to achieve adequate engine cooling. This configuration, identified as a "carburetor parts list," must be approved for the engine under Part 33 and should be listed on the type data sheet for the rotorcraft.

AC 27.901A. § 27.901 (Amendment 27-23) INSTALLATION.

a. Explanation. Amendment 27-23 changes § 27.901(b)(1) to require a satisfactory determination that rotorcraft can operate safely throughout adverse environmental conditions such as high altitude and temperature extremes. This amendment was needed to provide consistent application of environmental qualification aspects. This amendment also added a new paragraph (§ 27.901(b)(5)) to require design precautions to minimize the potential for incorrect assembly of components and equipment essential to safe operation.

b. Procedures. All of the policy material pertaining to this section remains in effect with the addition of design precautions. Design precautions should be taken to minimize the possibility of improper assembly of components essential to the safe operation of the rotorcraft. Fluid lines, electrical connectors, control linkages, etc., should be designed so that they cannot be incorrectly assembled. This can be achieved by incorporating different sizes, lengths, and types of connectors, wires, fluid lines, and mounting methods.

AC 27.903. § 27.903 (Amendment 27-11) ENGINES.

a. Engine Type Certification.

(1) Explanation. Section 27.903(a) is intended to ensure that engines used in type certified aircraft are properly qualified and that the associated installation requirements are established.

(2) Procedures.

(i) Compliance can be documented by verification that a type certificate data sheet has been issued by the FAA/AUTHORITY for the engine identified by the rotorcraft manufacturer as the engine planned for use in the rotorcraft. Reciprocating engines must have been qualified to a special test plan (§ 33.49(d)) to be eligible in rotorcraft. This eligibility should be verified by a note on the engine type certificate data sheet.

(ii) On some occasions, the engine certification program is conducted concurrently with the rotorcraft certification program. This is technically acceptable provided the engine type certificate is issued prior to the rotorcraft type certificate. However, practical considerations involving the use of unapproved engine installation data and the probability of engine design changes during the engine certification program that impact the rotorcraft certification program dictate that special procedures must be introduced to assure that the final rotorcraft certification program is satisfactory. If the engine under consideration is merely a minor model change from a previously certificated engine and these changes are unlikely to cause rotorcraft certification problems and do not involve significant installation aspects, the rotorcraft project engineer need only to follow the engine certification program by routine checks with the

FAA/AUTHORITY office responsible for engine certification and, as a final pre-type certification item, verify that the engine type certificate has been issued. Rotorcraft Type Board agenda/minutes should reflect the ongoing status of the engine TC program. For rotorcraft certification programs involving new or significantly changed engines, the powerplant certification engineer for the rotorcraft should become as familiar with the engine as practicable with particular attention to engine ratings, limitations, performance, engine/rotorcraft interface aspects, and any Part 27 certification requirement involved in the engine program (fuel/oil filters, fuel heaters, integral firewalls, etc.) and establish an appropriate working arrangement with the FAA/AUTHORITY engine certification office to monitor changes in the engine certification progress which may impact the rotorcraft certification program. In addition, any rotorcraft certification activity such as test plans, analysis, compliance inspections, etc., which involves the engine should be accepted on a conditional basis; i.e., pending confirmation of completion of the engine program without changes pertinent to these aspects of rotorcraft program. The rotorcraft applicant should be advised of any limitations in this procedure, and that normally, the engine certification program should be complete before authorizing formal FAA/AUTHORITY participation in the rotorcraft certification plan; i.e., TIA.

b. Engine cooling fan protection.

(1) Explanation. Section 27.903(b) is intended to provide safety to the rotorcraft in the event of an assumed cooling fan blade failure or to prescribe a test to show that the cooling fan blade retention means is sufficient that blade failure is not a consideration.

(2) Procedures. The applicant may select § 27.903(b)(1), (b)(2), or (b)(3) to show compliance with this section. If § 27.903(b)(1) is selected, a demonstration should be conducted to show that at the maximum fan speed to be expected, a failed blade is contained within a housing or shroud which is included in the proposed type design and designated by the applicant as the containment shield. The rotational speed required may be related to an overspeed limiting device or to the maximum transient speed to be expected from analysis or test of the system or component which drives the fan. For components driven directly by the engine, output shaft disconnect and the subsequent terminal speed of the engine may set the test condition. To conduct an overspeed blade failure containment demonstration, applicants have found it convenient to progressively weaken a blade to induce failure at or above the required demonstration speed. Blade failure may be expected to subsequently fail some or all of the remaining blades. This condition, provided all blades are contained, is acceptable for showing compliance with this rule. However, the corresponding loss of cooling may be unacceptable if it causes the loss of any function essential to a controlled landing.

(3) Section 27.903(b)(2) may be selected; however, without containment, damage to any component or structure in the plane of the fan rotor or any other trajectory to be expected should not cause the loss of any function essential to a controlled landing.

(4) If § 27.903(b)(3) is selected, a spin test at 122.5 percent of the maximum speed associated with either engine terminal speed or an overspeed limiting device would be acceptable to show compliance. No failure should occur and distortion should not result in fan element contact with housings or other adjacent components. (Note: 150 percent of the centrifugal force is achieved at 122.5 percent of the rotational speed.)

c. Turbine Engine Installation.

(1) Explanation. The certification of turbine engines and particularly, the qualification of turbine rotors, assumes that the limitations established during these certifications will be accurately and rigorously observed during ground and flight operations in an aircraft. This paragraph is intended to promote this concept.

(2) Procedures. Primary engine limitations in the form of time, gas temperature, torque, and rotational speed and their corresponding allowable transient values are defined in the approved engine installation manual. The rotorcraft manufacturer must provide reliable, accurate means to assure that these limitations are not exceeded. These means may be in the form of automatic limiters or by crew monitoring of appropriately marked instruments. The FAA/AUTHORITY powerplant certification engineer and the rotorcraft manufacturer's staff should verify these aspects by:

(i) Evaluating all applicable instrument, indicator, or warning devices, including transmitters, and limiting devices, if any, for system tolerances.

(ii) Closely reviewing the component qualification reports of items in 398c(2)(i) above to verify that these devices are properly qualified and that any deviations are acceptable.

(iii) Assuring that maintenance data is provided for functional checks and calibration of instruments and devices which are used to monitor or protect critical turbine rotor limitations. Preflight checks for automatic limiter devices may be appropriate.

(iv) Verifying that instrument markings are clear and relatively simple, that corresponding flight manual instructions and descriptions are straightforward and complete, and instruments are located and orientated to minimize the probability of reading error.

AC 27.903A. §27.903 (Amendment 27-23) ENGINES.

a. Explanation. Amendment 27-23 adds a requirement to § 27.903(a) that requires reciprocating engines used in rotorcraft to be certified in accordance with the special rotorcraft engine test requirements in § 33.49(d). This change was needed to ensure that certification requirements are not overlooked when reciprocating engines are installed in rotorcraft to be certified under Part 27 requirements. Section (b) was revised to prescribe tests and qualifications for powerplant area cooling fans. This rule change requires cooling fans to be designed and installed to enable safe landing of the rotorcraft following a fan blade failure. Compliance with the previous requirements could result in hazards to the rotorcraft with the loss of cooling air to critical powerplant components. A new paragraph was also added to the rule for cooling fans, which are part of the powerplant installation. It should be determined that no cooling fan blade resonant conditions exist within the operating limits of the rotorcraft unless a fatigue evaluation is conducted. These requirements will ensure that correct qualification procedures are used for rotorcraft engines and that all powerplant cooling fans are properly tested.

b. Procedures.

(1) Engine type certification. All engines installed in rotorcraft should have a type certificate. The specific certification requirements for reciprocating engines when installed in rotorcraft are found in the paragraph listed in Part 33. Engines certificated under other approved certification rules (CAR Part 13 and FAR § 21.29, for imported engines) are also eligible. If a component, system, or arrangement is certified under Part 33 or other requirement, the applicant is not relieved of the necessity to comply with the requirements of Part 27. If the component, system, or arrangement, supplied as a part of a certificated engine, meets the Part 33 and Part 27 requirements, subsequent changes to these components, systems, or arrangements could negate compliance with Part 27.

(2) The applicant may select §§ 27.903(b)(1)(i), (b)(1)(ii), or (b)(1)(iii) to show compliance with this section.

(i) For compliance with § 27.903(b)(1)(i), a demonstration should be conducted to show that at the expected maximum fan speed, a failed blade will be contained within a housing or shroud that is included in the proposed type design and designated as the containment shield. The maximum fan rotational speed may be related to an overspeed limiting device or to the expected maximum transient speed from analysis or test of the engine, system, or component which drives the fan. For fans driven directly by the engine, output shaft disconnect and the subsequent terminal speed of the engine may establish the maximum fan speed for the test condition. To conduct an overspeed blade failure containment demonstration, applicants have found it convenient to progressively weaken a blade to induce failure at or above the required demonstration speed. Blade failure may be expected to subsequently occur on some or all of the remaining blades. This condition, provided all blades are contained, is acceptable for showing compliance with this rule. However, the corresponding loss of cooling may be unacceptable if it causes the loss of any function essential to continued safe flight and landing.

(ii) For § 27.903(b)(1)(ii) compliance, if containment protection is not installed, damage to any component or structure within the trajectory of the failed fan rotor should not cause the loss of any function essential to a controlled landing.

(iii) For § 27.903(b)(1)(iii) compliance, a spin test should be conducted. For fans driven directly by the engine, the test should be conducted at 122.5 percent of the terminal engine rotational speed that will occur under uncontrolled conditions, or at 122.5 percent of the maximum engine rotational speed that would be controlled by a reliable, approved engine overspeed limiting device. For fans driven by the rotor drive system, the test should be conducted at 122.5 percent of the maximum rotor drive system rotational speed expected in service, including transients.

(Note: Capability to withstand the ultimate load of 1.5 times the centrifugal force means that no failure should occur and distortion should not result in fan element contact with housings or other adjacent components during the 122.5 percent spin test which equates to 150 percent centrifugal force).

(3) Fatigue. If the cooling fan is not included in the fatigue evaluation under § 27.571, it should be shown that the cooling fan blades are not operating at resonant conditions within the normal operating limits of the rotorcraft.

AC 27.907. § 27.907 ENGINE VIBRATION.

a. Explanation. Section 27.907 is intended to require the design of the rotor drive system, including the engine, to be free from harmful vibration. A vibration investigation is required.

b. Procedures. Review Order 8110.9, Handbook on Vibration Substantiation and Fatigue Evaluation of Helicopter and other Power Transmission Systems. Note that the mechanical coupling of the engines to the rotor drive system creates, for torsional vibration considerations, one, rather complicated, drive system which responds to any forced or resonant frequency. Antinodes or nodes and frequencies may exist in the engine shaft which are absent when the engine is operated on a test stand; therefore, the vibration investigation conducted under Part 33 is not conclusive with respect to torsionals. As noted in Order 8110.9, the engine manufacturers' assistance is necessary to find compliance. Section 27.571 was amended by Amendment 27-12 to include "rotor drive systems between the engines and the rotor hubs" as part of the flight structure. This rule supplements § 27.907 and requires coordination with the structures certification engineer to avoid duplication of effort by the rotorcraft manufacturer. Advisory Circular 20-95, which provides acceptable methods of compliance with § 27.571, may also be used to find compliance with § 27.907.

In addition to basic drive system components such as main and auxiliary rotor drive shafts, the vibratory evaluation should include couplings, gear teeth, gear cases and splines, and should consider, where appropriate, low cycle fatigue associated with ground-air-ground cycles.

SUBPART E - POWERPLANT

ROTOR DRIVE SYSTEM

AC 27.917. § 27.917 (through Amendment 27-11) DESIGN.

a. § 27.917(a):

(1) Explanation. This paragraph requires the design of the drive system to include a means to automatically disengage the engine(s) from the rotor drive system in order to prevent excessive drag from an inoperative engine from adversely affecting the performance of the rotor system.

(2) Procedures. The design objective usually is met by installing a freewheeling or overrunning clutch in the drive shaft between the engine and the first part of rotor drive system. If lubrication for these clutches is required, it should be provided by a means that continues to function after an engine is made inoperative except that for single-engine rotorcraft, clutch lubrication need only be provided for autorotation descent with the engine inoperative. A 15-minute demonstration of freewheeling or overrunning operation is usually acceptable.

b. § 27.917(b):

(1) Explanation. This paragraph requires that control rotors (tail rotors, for example) will continue to be driven by the main rotors when the rotorcraft is in autorotation.

(2) Procedures. Provide hard mechanical interconnect shafting between the rotors such that the main rotor will drive the control rotor (tail rotor). Note that this requirement must be met with all engines inoperative, thus, the driving force for the tail rotors must depend on the autorotative driving forces inherent in the main rotor(s).

c. § 27.917(c):

(1) Explanation. This paragraph pertains to any device or feature designed into the rotor drive system intended to prevent damage in the event of excessive torque in the rotor drive system from high engine power or mechanical interference with normal rotation of the rotor drive system. The rule prohibits location of these devices in any part of the rotor drive system that is required to continue functioning to provide control of the rotorcraft.

(2) Procedures. Review the arrangement of the rotor drive system to determine that any intentionally designed weak links in the system, such as shear sections or slip clutches installed to relieve high torsional loads, are located so that their function will not

compromise the interconnect mechanism between the main and auxiliary rotors of the rotorcraft.

d. § 27.917(d):

(1) Explanation. This paragraph sets forth a definition of the rotor drive system and its associated components.

(2) Procedures. Coordinate with other certification personnel to ensure that other rules pertaining to rotor drive systems are properly addressed.

AC 27.921. § 27.921 ROTOR BRAKE.

a. Background. Rotor brake safety requirements are intended not only to prevent adverse effects on aircraft performance due to brake drag but also to minimize the possibility of fire. These fires, caused by friction from a dragging rotor brake, have occurred both in flight and during ground operation with extremely hazardous consequences.

b. General. This rule requires (1) that any limitations on the use of the rotor brake must be established, and (2) that the control for the brake must be guarded to prevent inadvertent operation.

c. Limitations.

(1) The limitations on the use of the rotor brake should first be defined by the applicant and will normally consist of merely the maximum speed eligible for application of the brake. In some installations, other limitations associated with engine operation may be specified.

(2) Control guard mechanisms to prevent inadvertent operation may be conventional. A cockpit evaluation of the guard should be conducted by flight test personnel to affirm the function of the guard, that markings, if any, are adequate, and that both latched and unlatched positions of the guard do not interfere with other cockpit functions.

d. Other rules require both generalized and specific rotor brake qualification tests. However, some significant aspects of brake safety tests are listed below for reference.

(1) Routine application of the brake at shutdown during the endurance test of § 27.923 and during the function and reliability tests of § 21.35.

(2) Torsional vibration loads in the rotor drive system and oscillatory loads in the brake components during a critical brake engagement procedure should be determined with appropriate consideration in the fatigue evaluation for these

components. Brake engagements should be conducted with and without collective control displacement as authorized by the flight manual or a training manual.

(3) Brake component temperature measurements during a critical brake application in conjunction with an evaluation of the general brake compartment for compliance with § 27.863.

(4) Placards, decals, and flight manual limitations and instructions appropriate to operate the rotor brake safely.

(5) An evaluation for hazardous failure modes as required by § 27.1309(b). If the brake hydraulic system is integral with the rotorcraft hydraulic system, failure modes of pressure regulators and control valves will be of interest. Mechanical cams, calipers, and levers may be prone to seize or fail to release the brake due, in part, to corrosion and lack of lubrication to be expected when brake components encounter high temperature cycling.

e. Maintenance manuals should be checked for completeness in the areas of wear limits for both pucks and disks, for disk warp limits, and for defects which induce brake chatter. Also, maintenance data to check for proper function of pressure modulating/relief devices should be included since misadjustments of this device can amplify the stresses and temperatures in the system.

AC 27.923. <u>§ 27.923 (Amendment 27-12) ROTOR DRIVE SYSTEM AND CONTROL MECHANISM TESTS</u>.

a. <u>Explanation</u>.

(1) This section is intended to require demonstration that the rotor drive system, as defined in § 27.917(d), is capable of normal operation within the limitations proposed, without hazard of failure from excessive wear or deterioration due to mechanical loads. The basic test is not designed and should not be expected to demonstrate safety from oscillatory stresses normally investigated under §§ 27.571 and 27.907, although any data generated by these tests applicable to showing compliance with §§ 27.571 and 27.907 may be used. Some variations in the endurance test plan to generate data applicable to the vibration substantiation effort or other qualification aspects may be acceptable if the basic requirements of the endurance test are preserved.

(2) This rule requires a series of runs consisting of a 60-hour, 30-hour, and 10-hour run for a total of (at least) 100 hours of testing, not including time required to adjust power or to stabilize operating conditions for those conditions that require stabilization. Extension of the total test time beyond 100 hours (or extension of any test run segment beyond the minimum) will occur if qualification for the 2 ½-minute one-engine-inoperative (OEI) optional rating is proposed by the applicant. The 30-minute OEI rating qualification test will extend the test beyond 100 hours for rotorcraft equipped with three or more engines.

(3) Section 27.923(b) requires the test to be conducted “on the rotorcraft.” This means a rotorcraft in conformity to the design for which approval is requested. However, many nonconformity features, such as doors, some cowling and instrumentation, fuel tanks (alternate external fuel supply may be utilized), interior features, fire detectors, extinguishers, inlet ducts, exhaust baffles, etc., may be acceptable provided each item is technically considered and found to be unimportant to the test results. Any significant deviations from the conformed rotorcraft configuration, such as using ground or flight test facilities instead of the rotorcraft, providing the conditions which exist on the rotorcraft can be accurately duplicated, should be defined in the test proposal and approved by the cognizant FAA/AUTHORITY engineering staff. The restraint (tie-down) arrangement used during the test will necessarily be arranged to react rotor thrust loads in lateral as well as vertical directions. However, the restraint should permit normal deflections due to rotor thrust in the engine and drive system support arrangement. Safety cables may be installed normal to the tailboom at the tail rotor gearbox location; however, restraint may be provided to keep airframe deflections from exceeding those expected in normal and accelerated flight.

(4) The test torque requirements of § 27.923 mean the torque values for which approval is requested, but must not exceed the values approved for each respective limit for the engine being used. However, an applicant should be allowed to qualify the rotor drive system for torque values higher than those for which approval is requested if the engines actually used are capable of the torque and can be shown by an output shaft torsional investigation to be equivalent or conservative with respect to torsional vibration to the engines proposed for the initial certification configuration. Variations in rotational speed from the certification values should not be allowed except where careful evaluations of vibration aspects, bearing loads, centrifugal stiffening effects, and torque variations are conducted.

(5) The rotor configuration required by § 27.923(b) is intended to ensure that lift, torque, and vibration loads to be expected in service are introduced into the endurance test, although the presence of the vibration aspects does not normally satisfy the vibration evaluations required by §§ 27.571 and 27.907. In fact, vibration modes may be changed and amplified by the tie-down restraints and the increased thrust to be expected from in-ground-effects on the rotor system. These effects, although unquantified, are intended as a normal part of endurance testing. Preproduction rotor blades have been successfully used in endurance tests but only after specific investigations of blade properties such as stiffness, inertia and inertial distribution, thrust and blade bending, and torsional frequency response have been carefully compared to ensure validity of the test. The endurance test includes testing the rotor control mechanism. Conformity of the rotors may be very significant to this aspect of the test.

(6) For approved designs, some drive system changes or mechanical power increases may only require partial testing to satisfy § 27.923 requirements provided an equivalent level of safety finding can be made for the remaining requirements based on the previously approved data.

b. Procedures.

(1) Section 27.923(a) requires the rotor drive system and rotor control mechanism to be in a serviceable condition at the end of the test. Verification of this requirement requires a complete disassembly and examination of the entire rotor drive system and rotor control mechanism. The disassembly itself should be closely monitored for evidence of adequate breakaway torque on all bolted fasteners. Samples of lubrication from oil sumps and filters should be retained for spectrographic analysis, and seals should be examined for possible damage due to test requirements. Care should be taken to differentiate between seal damage and bearing damage due to disassembly procedures so that the direct results of the test may be properly considered. Close visual observation of each tooth on each gear is necessary to affirm proper load/contact patterns and absence of excessive surface stress or scrubbing motions. Bearings should be examined to verify that ball or roller paths are within limits, bearing cages are undamaged, and bearing balls or rollers and their races are free from pitting. Any evidence of bearing races turning or spinning in respective housing or bores probably indicates design or fit deficiencies. The applicant should have available wear limits data which include items such as distance across pins and tooth profile limits for gears. Many of these items require special, close tolerance inspection equipment and trained inspectors to determine compliance. In some instances, bearings, clutches, oil pumps, etc., should be returned to the original manufacturer for a finding of serviceability. Localized overheating usually exhibited by discolorations is an indication of an unsatisfactory condition. Should any of the items discussed above or other defects appear such that the component is unserviceable, a redesign which includes recognizable improvements should be required before authorizing a retest. To simply “try again” in hopes of success should not be accepted.

(2) Section 27.923(a) also prohibits intervening disassembly which might affect test results. Generally, this simply means no disassembly whatsoever. However, some very limited disassembly can usually be conducted provided care is used to ensure that items such as critical fastener torques or gear backlash controls are not disturbed.

(3) Section 27.923(b) requires that each rotor drive system and control mechanism be tested for not less than 100 hours. This endurance test is intended to demonstrate a minimum level of reliability and proper functioning of this system. The test should be conducted on the rotorcraft to provide the most realistic test environment. Exceptions can be made only if a ground or flight test facility is used that closely simulates the support and vibration conditions existing on the actual rotorcraft. The rotor system installed on the ground test article should be the same as that used on the flight test vehicle. If significant productivity changes are made after completion of the tests, retesting may be required.

(4) In § 27.923(c), (d), (e), and (f), the runs should be made with the proper torques, rotor speeds, and control positions specified by each paragraph. The controls discussed in each paragraph are the flight controls; i.e., cyclic and directional controls

for rotorcraft with tail rotor and single main rotor. The collective control is normally used to set power and is not involved in the control cycling described in § 27.923(h). During control cycling the controls may be cycled from stop to stop, or a limited travel may be accepted if the travel produces the maximum fore and aft, left and right, and yaw thrust components of the rotors as measured in flight for a particular flight condition. One method of determining the required control displacement is to measure main rotor mast bending in level forward flight at maximum continuous power (or the power associated with the maximum rearward flight speed to be expected) for the aft control displacement limit. Using the same mast bending instrumentation with the rotorcraft in the ground tie-down situation and with collective control set for maximum continuous power, displace the cyclic fore and aft to obtain the same mast bending as measured in flight. Similar measurements and control displacements may be used for sideward thrust components. Yaw control displacement should consider maneuver requirements in conjunction with sideward flight. Critical gross weight and center of gravity should be used to establish test conditions. Vertical thrust may be used during the takeoff run and the runs at 2 ½-minute power. OEI runs should be conducted with the cyclic set for maximum forward thrust for the 30-minute power run and at maximum vertical thrust for the 2 ½-minute power run. For these runs and any run that does not specify the position for the yaw control, that control should be set to react main rotor torque.

(5) Section 27.923(e) prescribes the takeoff portion of the endurance test. This is a 10-hour test that must be run at not less than the maximum torque and the maximum RPM to be approved for takeoff. For this test the main and auxiliary rotor controls should be in the normal position for vertical ascent. If the applicant elects (for a multiengine rotorcraft) to perform the 2 ½-minute OEI power test, a series of three 2 ½-minute repetitive runs should be conducted during the course of the 10-hour test. For these tests, main and auxiliary rotor controls should be in the position for vertical ascent, and power settings for the operating engine should provide red line torque (or manifold pressure) and RPM for the 2 ½-minute OEI power rating. The nonoperating engine may be allowed to operate at idle or may be shut down.

(6) The torque and speed requirements in § 27.923(e) for the optional 2 ½-minute OEI tests should be interpreted as described above for the takeoff runs. If the test is conducted during warm ambient conditions, excessive engine gas temperatures may be required to achieve the torque and speed conditions required by this part of the test. Minor adjustments in the run schedule may be allowed to take advantage of cooler nighttime ambient temperatures. Addition of water/alcohol systems to increase engine hot-day power may be appropriate in some instances. Liquid nitrogen spray into engine inlets has also been used to depress inlet temperatures sufficiently to obtain test conditions.

(7) The requirement in § 27.923(g) for declutching the engine may be difficult to achieve if engine decelerations and rotor system decelerations rates are similar. In some cases, the engine fuel control deceleration schedule may be adjusted to achieve clutch disengagement; otherwise, an engine shift brake mechanism may be needed.

(8) Tests described in § 27.923(h) should be conducted under the conditions of maximum continuous power and RPM as described in § 27.923(c).

(9) Section 27.923(i) requires 200 clutch engagements. This test is prescribed to establish a level of reliability of clutch components installed as a part of the rotor drive system of rotorcraft. The clutch tests apply to all clutches installed to comply with § 27.917(b), and each such clutch must be tested. A rotor brake is not required for certification, although a brake of some type may be installed temporarily to facilitate conducting the clutch testing required by this section. Clutch disengagement is also required by this section; thus, malfunction of the disengagement feature would be a basis for discontinuance. Some rotorcraft configurations (those with single-spool turbine engines or reciprocating engines) include an additional clutch to decouple the engine from the drive system to facilitate engine starting. These clutches should also be exercised at least 200 times during this test.

(10) Section 27.923(j) sets forth the optional tests to be conducted if a 30-minute OEI rating is requested. Flight control positions should be set for level flight or climb, whichever produces the maximum forward thrust component, and the antitorque system control should be set to react the maximum rotor torque. The torque and rotational speed values should be the maximum for which approval is requested.

c. Additional Test Considerations.

(1) Pressure Lubricated Gearboxes. The endurance test hardware can be adjusted/modified to sustain high-limit oil temperature and low-limit oil pressure to provide a basis for approval of the values listed as limits. A minimum of 20 hours at maximum continuous torque and maximum continuous rotational speed should be involved in the test. Other parameters such as minimum oil temperature and maximum oil pressure may more appropriately be evaluated by bench test. The significant points here are effects of extremely high oil pressure (due to the high viscosity of cold oil) on any positive displacement oil pump, on filters for possible collapse, on oil coolers for possible rupture due to internal pressure, seals, bypass valves, and most important, adequate lubrication of gears, bearings, etc., under conditions of minimal oil flow. Normally, an operational restriction against exceeding idle power/speed conditions until significant warm-up occurs is prescribed. Individual component qualification tests may provide data to meet some of these aspects.

(2) Asymmetric Power Inputs. The existing endurance test schedule does not necessarily provide for any asymmetric power inputs from multiengine drive system arrangements. For this situation, the drive system should at least be subjectively evaluated for possible hazards or excessive loads to be expected from asymmetric torque inputs. If required, additional testing should be considered.

(3) Accessory Drives. Normally, all accessory drives on a gearbox will be loaded during the endurance test. Electrical load banks or other suitable methods may be used to ensure that the generator drives are loaded and thus properly qualified.

Hydraulic pumps may be loaded by resetting hydraulic system relief valves to maintain limit pressure (load) continuously. If this condition is excessively severe, a method of load cycling may be appropriate. Note that accessory loads reduce the power available to the main rotor. Also, tail rotor loads are, insofar as the transmission is concerned, another large accessory. Care should be taken to ensure that in-flight unloading of these accessory drives, including the tail rotor, does not subject the main gearbox to loads significantly beyond those qualified by endurance tests.

(4) Gearbox Oil Tanks. Normally, gearbox oil is contained in an integral cast sump which, for other reasons, has sufficient strength to obviate the need for pressure tests. However, a subjective evaluation should be made to ensure that detail design features such as sight gauges, filler caps, etc., offer adequate strength.

AC 27.923A. § 27.923 (Amendment 27-23) ROTOR DRIVE SYSTEM AND CONTROL MECHANISM TESTS.

a. Explanation. Amendment 27-23 revised § 27.923(c) to remove the references to “engine power” to avoid confusion. Previous wording could have been interpreted to mean tests prescribed by this section should be conducted at powers corresponding to engine ratings established under Part 33, rather than rotorcraft powers which may be lower than those established under Part 33, but selected by the applicant as a limit on their product. Section 27.923(d) was revised to remove the references to “engine power” and to clarify the test requirements for 30-minute and continuous OEI powers. Previously, §§ 27.923(e) and (j), as they relate to the 2 ½-minute and 30-minute power ratings, respectively, provided for only minimal testing of the capability of the rotor drive system to sustain these powers. Amendment 27-23 amended these sections to extend the testing to adequately assure valid qualification tests. These changes ensure the integrity of the rotor drive system so that it will safely sustain the higher stresses expected with actual, repeated use of these power ratings. A new § 27.923(k) was added that provides a qualification test schedule for the new, optional, continuous OEI rating.

b. Procedures. The policy material pertaining to this section remains in effect with the following additions:

(1) Section 27.923 requires a minimum of 100 hours of endurance testing.

(i) For single engine rotorcraft and others that will not have OEI ratings, the 100-hour test is comprised of 60 hours at not less than maximum continuous power, 30 hours at not less than 75 percent maximum continuous power, and 10 hours at not less than takeoff power.

(ii) For multiengine rotorcraft for which OEI ratings are requested, the test is comprised of 60 hours at not less than maximum continuous power, 25 hours at not less than 75 percent maximum continuous power, 10 hours at not less than takeoff power, and 5 hours at simulated OEI power conditions.

(2) The endurance time, cited in paragraph b(1) above, excludes the time required to adjust power or to stabilize operating conditions. Extension of the total test time beyond 100 hours (or extension of any test run segment beyond the minimum) will occur if qualification for the 2 ½-minute, 30-minute, or continuous OEI optional ratings is proposed by the applicant for rotorcraft equipped with two or more engines.

(3) The requirements in § 27.923(f) stipulate that the endurance tests conducted at maximum continuous and 75 percent maximum continuous power should be conducted in intervals of not less than 30 minutes. These tests may be conducted on the ground or in flight. The takeoff power endurance test described in § 27.923(e) should be conducted in intervals of not less than 5 minutes.

(4) The new § 27.923(k) sets forth the tests to be conducted if a continuous OEI rating is requested. Flight control positions should be set for level flight or climb, whichever produces the maximum forward thrust component. The anti-torque system control should be set to react the maximum rotor torque. The torque and rotational speed values should be the maximum for which approval is requested.

AC 27.923B. <u>§ 27.923 (Amendment 27-29) ROTOR DRIVE SYSTEM AND CONTROL MECHANISM TESTS</u>.

a. <u>Explanation</u>. Amendment 27-29 added § 27.923(e)(2) that defines qualification tests for 30-second/2-minute OEI ratings. This new paragraph also allows for the 30-second/2-minute OEI portion of the endurance test to be accomplished on a representative bench test facility using the drive system components which can be adversely affected by these tests.

b. <u>Procedures</u>.

(1) For accomplishment of the endurance test for 30-second/2-minute OEI, § 27.923(e)(2) requires that 10 applications of 30-second/2-minute OEI power be demonstrated for each power section during the 10 hour takeoff power segment of § 27.923(e). Each 30-second/2-minute OEI application should be conducted immediately following a 5 minute stabilized takeoff power run. Following the 5 minute takeoff power run, one engine must simulate a power failure and each engine providing power after the failure must apply the maximum torque and maximum speed for use with 30-second OEI power. This power level should be maintained for at least 30-seconds. The 30-second OEI power should then be followed by an application of the maximum torque and maximum speed for 2-minute OEI power for at least 2 minutes. Section 27.923(e)(2) also requires that one of the 30-second/2-minute OEI segments for each engine be accomplished from the flight idle condition.

(2) Additionally, due to the damage inflicted on the engines and the ensuing cost caused by operating the engine at these powers, the 30-second/2-minute portion of the endurance test can be accomplished on a bench test found to be representative of

the rotorcraft. The representative bench test rig should have the ability to generate the torques, speeds, vibration frequency, and acceleration rate generated by the rotorcraft. The power should have the same method/path of application as that used on the rotorcraft. The test rig should be configured with the same components used for conducting the endurance test on the rotorcraft except that the test components not affected by asymmetric power application may not have to be installed (i.e., if a combining gearbox is used it may not be necessary to have the main transmission installed on the bench test rig).

(3) The takeoff portion of the endurance test should be accomplished on the rotorcraft. When conducting the bench test for 30-second/2-minute OEI it is not necessary to repeat the takeoff portion of the endurance test; however, the simulated power failure and application of 30-second/2-minute OEI power by the remaining engine(s) should be accomplished after the input power has stabilized at takeoff power.

AC 27.927. § 27.927 (Amendment 27-12) ADDITIONAL TESTS.

a. Section 27.927(a):

(1) Explanation. This paragraph is the authority to require any special tests or investigations to establish that the rotor drive system is safe.

(2) Procedures. The certification engineer should review the design of the rotor drive system and its installation and intended operation for features or conditions that may not be adequately qualified in the tests prescribed by this part and, if necessary, additional qualification test programs should be developed and accomplished to ensure safe operation of the rotor drive system. Items of interest would include poorly defined load paths associated with redundant design features, flight deflections of structure and of mounting arrangements, and special or unusual operating procedures which may be anticipated or proposed by the applicant.

b. Section 27.927(b):

(1) Explanation. This paragraph prescribes testing to qualify the rotor drive system for the power excursions to be expected with governor-controlled engines wherein the power from the engine(s) changes automatically to maintain rotor speed at preselected values. At high collective flight control displacements, the normal rotor speed droop will result in the governor-controlled engine(s) automatically accelerating to maximum fuel flow or to any other power, speed, temperature, or torque limiting device, regardless of crew action or artificially established limitations reflected by instrument markings. This high power condition can occur typically during a normal landing when the crew applies high collective to cushion ground contact or, for multiengine rotorcraft, during any flight regime when an engine fails and the corresponding loss of power results in drooping the rotor speed. Special tests are prescribed by this section to provide assurance that the rotor drive system can safely sustain these conditions. The tests of this section should be conducted without intervening disassembly, and all rotor

drive system components should be in serviceable condition after the test. It is permissible but not required that these tests be performed on the same specimen of the rotor drive system used to show compliance with § 27.923.

(2) Procedures. Testing as prescribed by this section should be conducted on a ground-test rotorcraft conformed to the type design suggested for the endurance test of § 27.923. In most cases, testing to comply with § 27.927(b)(1) is accomplished as a continuation of the test of § 27.923 using the same test vehicle. For this test, the main rotor control (cyclic/collective) may be set to simulate vertical lift. The auxiliary rotor control (antitorque) may be set or adjusted to react main rotor torque. Rotation speed should be maximum normal for the test condition; i.e., all engines operating as for takeoff. Using the collective control, obtain torque as required to meet either § 27.927(b)(1)(i) or (ii). This will normally be 110 percent of takeoff torque or a lower value as limited by an approved, reliable device to simultaneously limit torque on all engines. If individual torque limiters are provided for each engine, rigging tolerances should be at maximum allowed mismatch for the type design. For the one-engine-inoperative (OEI) test of § 27.927(b)(2), rotor RPM droop, if any, may be allowed as would occur in service. Since this OEI test requires the remaining engine(s) to produce power not usually available under normal atmospheric conditions, supplemental power augmentation may be needed such as inlet air refrigeration, ramming, or overfueling the engine. Alternatively, bench testing with a transmission test rig may be appropriate providing close simulation of the drive system torsionals, shaft/coupling, misalignment, etc., is achieved. Overtesting (excessive torque) to compensate for inadequacies in the bench test may be negotiated with the FAA/AUTHORITY approval office. Note that compliance with § 27.903(b) requires that the remaining engine(s) be capable of safe, continued operations under the high power conditions of this test. This may require the engine manufacturer to conduct special testing or to produce suitable evidence that the stresses (and gas temperatures) associated with these governor-induced high power excursions do not compromise the airworthiness of the remaining engines or their capability to produce topping power automatically during the initial moments of flight after an engine failure.

c. Section 27.927(c):

(1) Explanation. This paragraph prescribes a test which is intended to demonstrate that in the event of a pressure failure of any pressurized lubrication system used on the rotor drive system, no failure or malfunction will occur in the rotor drive system that will impair the capability of the crew to execute an emergency descent and landing. The lubrication system failure modes of interest usually are limited to failure of external lines, fittings, valves, coolers, etc., of pressure lubricated transmissions.

(2) Procedures. Conventionally, a bench test (transmission test rig) is used to demonstrate compliance with this rule. Since this is essentially a test of the capability of the residual oil in the transmission to provide limited lubrication, a critical entry condition for the test would be the critical eligible lubricant preheated to the transmission oil temperature limit. With the transmission operating at maximum normal speed, with

lubricant as described above, with nominal cruise torque applied (reacted as appropriate at main mast and tail rotor output quills), and with a vertical load at the mast equal to gross weight of the rotorcraft at 1g, disconnect or cause to leak an external oil plumbing device. Upon illumination of the low oil pressure warning (required by § 27.1305), reduce engine input torque to zero to simulate autorotation, and continue rotation for 15 minutes. Apply input torque to simulate a minimum power landing for approximately 15 seconds to complete the test. Successful demonstration may involve limited damage to the transmission provided it is determined that the autorotative capabilities of the rotorcraft were not significantly impaired.

AC 27.927A. § 27.927 (Amendment 27-23) ADDITIONAL TESTS.

a. Explanation. Amendment 27-23 changed § 27.927 by adding a requirement that the rotor drive system overtorque tests prescribed by § 27.927(b)(3) be conducted at the maximum rotational speed intended for the power condition of the test. The previous rule only specified the torque to be applied to the rotor drive system during the overtorque test.

b. Procedures. The changes to this section did not change the suggested method of compliance.

AC 27.931. § 27.931 SHAFTING CRITICAL SPEEDS.

a. Explanation.

(1) At certain speeds, rotating shafts tend to vibrate violently in a transverse direction. These speeds are variously known as “critical speeds,” “whirling speeds,” or “whipping speeds.” The vibration results from the unbalance of the rotating system and can be shown to reach destructive values with only minimal unbalance. The nature of this phenomena is that as shaft rotational speed increases, residual unbalance in the shaft gives rise to centrifugal forces. These forces cause the shaft to rotate in a bent or bowed configuration with the centrifugal force induced bending loads being balanced by coriolis and elastic forces in the shaft. As shaft rotational speed increases, the centrifugal forces increase to the point at which they exceed the elastic forces in the shaft, and divergence occurs. This point in the speed range is called the critical speed. At shaft speeds above the critical speed, a 180° phase change occurs, the shaft’s mass center moves toward the center of rotation, and the amplitude of vibration diminishes with further increases in shaft speed.

(2) A design option would be to operate the shafting subcritical; i.e., below the first critical speed, with adequate margins between critical speed and the maximum allowable speed, including transients. However, another option, that of supercritical shaft operation; i.e., operating above the first or even higher critical speeds with adequate margins between any critical speed for the normal operating speed range may be permitted. This latter option requires some form of system damping to permit safe

transition through the critical speed range and to avoid excessive nonsyncronous vibrations or instability when transitioning through the critical speed.

(3) A review of typical design practices and drive system arrangements discloses several types of shaft support and loading:

(i) Main rotor/mast/transmission assemblies rigidly mounted to the airframe.

(ii) Main rotor/mast/transmission assemblies compliantly mounted to the airframe.

(iii) Main rotor supported through a bearing arrangement by a rigid nonrotating structure with a coaxial torque shaft driving the rotor.

(iv) Cross-shafting, interconnect shafting, tail rotor drive shafting which are generally supported by gearboxes at each end and by hanger bearings and couplings at intervals along the tail rotor drive shaft.

(v) Engine to transmission shafting which, for compliant pylons, incorporates a flexible or geared coupling, to accommodate the misalignment and chucking.

(vi) Tail rotor/mast/gearbox supported on the tailboom or near the upper extremity of a vertical fin.

(4) With regard to compliant pylon mountings, recent developments in vibration control have led to rotor isolation wherein the fuselage is isolated from the rotor and transmission, resulting in improved vibration and system reliability. Rotor isolation systems typically entail the installation of isolation devices at the transmission-airframe interface. The crux of rotor isolation is providing adequate, low-frequency isolation without excessive relative displacement or loss of mechanical stability. Rotor isolation affects shaft critical speeds in the following ways:

(i) First, the transmission mounting configuration, system stiffness, and tuning requirements may result in different fore-and-aft and lateral natural frequencies, imposing additional analytical requirements. For compliant mounting, the response while transitioning through the fundamental or rocking modes is generally controlled by dampers or elastomeric elements.

(ii) Second, the relatively high displacements permitted by the isolation system, depending on configuration, may result in variations in shaft misalignment and length thus adding further complexity to the analytical prediction of critical speeds.

b. Procedures.

(1) Subcritical Shafting Designs. Three basic methods of qualification may be considered, with the required margins relative to the degree of assurance provided. The margins are shown for guidance only.

(i) Analytical.

(A) Simplistic model(s) as shown in figures AC 27.931-1 and AC 27.931-2; 35-percent margin shown above maximum operating speed.

(B) Detailed model, taking into account significant variations in shaft stiffness, mass distribution, cone adapters, support bearing stiffnesses, support structure; 20-percent margin shown above maximum operating speed.

(ii) Analytical supported by tests. Analysis supported by shake test (rotating or nonrotating) or by bench test, where appropriate adjustments are made for differences between the bench and the aircraft; 15-percent margin shown above maximum operating speed.

(iii) Whirl test on the aircraft.

(A) For all cases, it should be shown that, under maximum permissible unbalance and at the maximum operating speed, the shafting and support structure has acceptable clearance and does not have excessive vibration.

(B) For compliant pylon mountings, damping of the rigid body rocking modes, which are often transitioned during runup to normal speed (and which are not critical flexing modes), may be verified by analyses, laboratory tests, or ground runup with the rotor at maximum permissible unbalance. Damping on the order of 5 percent equivalent viscous damping is generally acceptable.

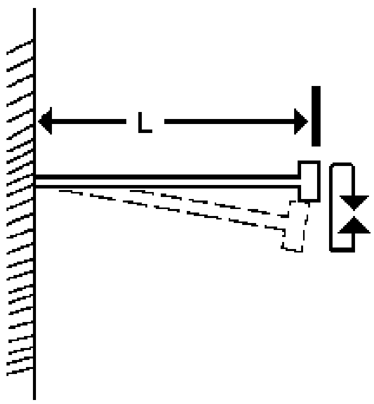
(C) For tail rotor masts, the analysis should include fixed system structural response including tailboom, fixed control surfaces, and vertical fin. The frequency analysis will then contain both fixed system and rotating system modes. An energy analysis can then be used to identify whether the modes are predominantly fixed system or rotating system modes. Systems with up to 35-percent energy in the rotating system have been operated in the field without significant problems. For this type of shafting installation, it is advisable to avoid fixed system modes at multiples of shaft speed, particularly where highly non-isotropic mountings exist.

(2) Supercritical Shafting Design. Another facet occasionally encountered with shafting is the concept of normally operating at speeds above the critical speed, commonly referred to as “supercritical operation.” To function properly, suitable dampers must be installed to enable the shaft to pass safely through the lower critical speed up to the operating speed, and speed controls should be devised to avoid any tendency to operate continuously at any critical speed. Accurate balancing of the rotating components will also decrease the energy to be dissipated into the damping

device during transition thereby increasing its serviceability and reliability. Note that damper design and locations become more complex as selected operating speed increases through the third or fourth critical frequency. Multiple node points will exist where dampers will not be effective. Production specimen testing at high speed/high torque conditions should include checks for shaft straightness until experience verifies that shaft deflecting is not significant. For systems utilizing squeeze film dampers at the support bearings, variations in oil pressure, flow restrictions, and the effects of bearing preload should be evaluated. The effects of shaft and unbalance and the proximity of the damper to bottoming under maximum unbalance should be evaluated.

(3) If the shafting configuration of the rotorcraft includes universal joints or misalignment couplings, a velocity differential will exist across the joint which creates sinusoidal torques and bending moments at both shafts at multiples of the rotation speed. To avoid amplification of these torques and bending moments, the design should preclude coincidence of critical speeds and multiples of normal speeds.

(4) Order 8110.9, Handbook on Vibration Substantiation and Fatigue Evaluation of Helicopters and Other Power Transmission Systems, also addresses this subject. This document is distributed to section level and above in all Regional Aircraft Certification Offices.



$$W_{cr} = \sqrt{\frac{k}{M + 0.23m}}$$

W_{cr} = first critical speed, RAD/SEC

k = shaft spring rate, LB/IN = $3EI/L^3$

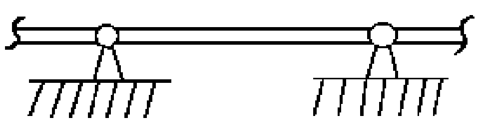
E = modulus of elasticity

I = moment on inertia

M = mass of weight, LB-SEC^2/IN

m = mass of shaft, LB-SEC^2/IN

FIGURE AC 27.931-1 CANTILEVERED SHAFT, FIRST CRITICAL SPEED



D/SEC

$$W_{cr} = a\sqrt{\frac{E\ I}{u\ L^4}}$$

W_{cr} = first critical speed-RAD/SEC

E = Young's modulus

I = inertia of shaft

u = mass per unit length

L = length between supports

a = a numerical constant: for first critical speed, $a = (pi)^2 = 9.87$

The numerical constant (a) for higher critical whirl modes or other shaft support systems may be derived from standard texts on this subject.

FIGURE AC 27.931-2 SHAFT BETWEEN SUPPORT BEARINGS, FIRST CRITICAL SPEED

AC 27.935. § 27.935 SHAFTING JOINTS.

a. Explanation. This rule requires the design of shafting joints to include provisions for lubrication when such lubrication is necessary for operation.

b. Procedures. Review the design of the rotor drive system for universal joints, slip joints (splines) and other shaft couplings. Lubrication access points (Zerk fittings) should be required unless the design incorporates alternate provisions for lubrication acceptable to the FAA/AUTHORITY and shown valid by test or experience.

AC 27.939. § 27.939 (Amendment 27-11) TURBINE ENGINE OPERATING CHARACTERISTICS.

a. Explanation. This section requires evaluation of engine operation, engine inlet airflow distortion, and engine/drive system torsional stability. A satisfactory rotorcraft design for all three items should be established by the manufacturer early in his development program since changes in design to satisfy these requirements are typically very expensive and will adversely impact other basic design features. The results of these evaluations are used to verify that FAA/AUTHORITY-approved Engine Installation Manual requirements are satisfied.

b. Procedures.

(1) Turbine engine operation.

(i) Explanation. Smooth, stable operation of turbine engines is essential to safety and control of rotorcraft. This can be adversely affected by rotorcraft maneuvers, turbulence, high altitude, temperature, airspeed, and installation features such as the engine air inlet duct, exhaust duct, and the location with respect to other airframe items which induce or influence air flow through the engine. Powerplant control displacement rate can also be a factor, although most modern engines incorporate internal protection for this aspect. The engine's tolerance to these factors is reflected as the "stall margin" which is established by the engine manufacturer through design and test. However, this stall margin is applicable only to an engine with a specified inlet and exhaust and at specified altitude, temperature, and effective airspeed. Typically, the specified engine inlet duct is a symmetrical bellmouth and the exhaust is a short straight duct of specified diameter and length. The stall margin, even under the above test conditions, usually varies with engine power, acceleration or deceleration, compressor air bleed, and accessory power extraction.

(ii) Procedure. The official flight test plan should include requirements to investigate the engine operating characteristics for stall, surge, flameout, acceleration and deceleration response, and transient response (within approved limits) throughout the operating range of the rotorcraft. This should include maximum airspeed-sideslip combinations, power recoveries, hover with wind from all azimuths and other maneuvers appropriate to the type. Recirculation of exhaust gases during hover can be critical for engine operation. Particular attention should be given to flight/operating conditions which can be judged critical from review of data on engine inlet pressure and temperature distribution patterns and engine stall margin data if available. High altitude has typically been critical for these tests and rearward flight at high altitude has resulted in unacceptable thermal distortions in the inlet due to reingestion. Stall, surge, or flameout which may be hazardous is unacceptable; i.e., causes loss of engine function, loss of control, severe torsional shock through the rotor drive system, or otherwise damages the rotorcraft.

(2) Vibration.

(i) Explanation. Engine airflow patterns are deflected or distorted by the presence of airframe inlet hardware, cowling, fuselage panels, and, to a degree, in almost all flight regimes. Additional items such as airframe installed particle separators, deflectors for snow, ice, or sand protection, and obstructions forward of the engine inlet, such as a hoist kit, could affect the engine air flow patterns. The rotating elements of the engine, particularly the compressor blades, will be subjected to a cyclically varying air flow as these elements move into and out of areas of deflected airflow to the engine. A corresponding aerodynamic load will be imposed on these engine elements. Since this loading is also cyclic, the possibility of critical frequency coupling with an engine component shall be investigated.

(ii) Procedure. Typically, this evaluation would involve installation in the engine inlet of a special multiple probe, total pressure sensing system, and flight testing which largely follows that prescribed for evaluation of engine operating characteristics as described above. Data from these tests can be reduced to create a pressure map at the compressor inlet face which, in conjunction with compressor speeds, may be used to determine the frequencies and relative amplitudes of the cyclic air loading imposed on the engine compressor blades. The engine manufacturer either supplies the sensing probe or specifies its design and performance. Also, the engine manufacturer may evaluate the test results or publish acceptance criteria. A wave analysis may be involved in identifying higher order excitations. Engine exhaust ducts which include bends, noise suppressors, or other obstructions may require an evaluation similar to that discussed above for the engine inlet. The engine manufacturer should be consulted for instructions or approval of this aspect. High performance engines may also require an engine inlet temperature survey. Details of instrumentation and acceptance criteria should be provided by the engine manufacturer. Engines equipped with only centrifugal compressors are less likely to encounter frequency coupling and may not require this investigation. The engine manufacturer’s recommendations should be followed in these cases.

(3) Torsional Stability.

(i) Explanation. Governor-controlled engines installed in rotorcraft are subject to a fuel control resonant feedback condition which could be divergent if not properly designed or compensated. This condition occurs when the response frequency of the governor on the engine is coincident with or close to a low order natural torsional frequency of the rotorcraft rotor drive system. Typically, these frequencies appear in the 3 to 5 CPS range. The manufacturer usually resolves torsional instability problems by introducing damping into the engine governor/fuel control. Provisions for this change must be supplied by or approved by the engine manufacturer. The final configuration may be a compromise between a lightly damped control, which will allow a positive but slow convergence of drive system torsional oscillations, and a highly damped control which exhibits excessive rotor speed droop or overspeed following rotorcraft collective control displacement.

(ii) Procedure. A ground and flight test program should be devised to evaluate the torsional response of the engine and drive system combination presented by the applicant. Instrumentation to record drive system torsionals should be applied to all major branches of the drive system. Engine parameters such as torque and power turbine speed should be recorded simultaneously with drive system parameters. The test program should include ground tie-down operation and flight operation across a range of engine power and rotor speeds while injecting control inputs as close to the first order drive system natural frequency as possible. Mechanical methods of making these inputs are not usually necessary if the desired frequency is in the 3 to 5 CPS range and the instrumentation readout confirms that the drive system was actually excited torsionally at its natural frequency. Control inputs should include collective, antitorque, and throttle. Also, cyclic inputs may be important on tandem rotor rotorcraft. The acceptance criteria may be dependent on several items. Among these are rotor and drive system fatigue loading, engine power response characteristics, limitations established by the engine manufacturer, etc. The acceptance criteria are usually stated as a percent damping (minimum). Typically, 1 percent of critical equivalent viscous damping (or greater) is acceptable. In effect, this means that the free vibration response to a control input damps to ½ amplitude in 11 cycles or less.

SUBPART E - POWERPLANT

FUEL SYSTEM

AC 27.951. § 27.951 (through Amendment 27-9) FUEL SYSTEM--GENERAL.

a. Explanation.

(1) The term "fuel system" means a system which includes all components required to deliver fuel to the engine(s). This includes, but is not limited to, all components provided to contain, convey, drain, filter, shutoff, pump, jettison, meter, and distribute fuel to the engines.

(2) Paragraph (a) of this section is a general statement of the performance requirements for fuel systems and constitutes authority to require the fuel system to be adequate notwithstanding compliance with detail requirements listed in §§ 27.953 through 27.999 of this subpart.

(3) Paragraph (b) of this section requires fuel systems to be designed so that air will not enter the system under any operating conditions by either arranging the system so that no fuel pump can draw fuel from more than one tank or by other acceptable means.

(4) Paragraph (c) of this section sets forth a fuel system performance requirement intended to ensure that ice to be expected in fuel when operating in cold weather will not prevent the fuel system from supplying adequate fuel to the engines. Although fuel system filters and strainers are the items in the fuel system most susceptible to clogging from ice particles in the fuel, this paragraph requires that the entire fuel system be shown to be capable of delivering fuel, initially contaminated with water and cooled to critical icing conditions, to the engine(s).

b. Procedures.

(1) For paragraph (a), the applicant should show compliance with the fuel system requirements of this subpart, except that if unusual fuel system arrangements or requirements exist which are not adequately addressed by these subparts, this paragraph may be used as authority to require special tests, analysis, or system performance needed for proper engine functioning.

(2) For paragraph (b), review the fuel system design with special attention to fuel tank selector valves, crossfeed systems, and multiple tank outlet arrangements to ensure that no fuel system configuration will allow air to enter the system. For questionable situations, the applicant should conduct ground tests and flight tests as necessary to verify compliance with this section.

(3) Paragraph (c) provides for sustained satisfactory operation of the fuel system, with initially ice-contaminated fuel. Since ice in the fuel system is not considered to be an emergency condition, but rather is an expected service encounter, compliance would not involve the imposition of special rotorcraft limitations. Flight manual instructions such as land as soon as practicable, reduce altitude to some value less than otherwise permitted, reduce power, turn on boost pumps, etc., are not appropriate in demonstrating compliance. Some methods of fuel system ice protection which have been used to show compliance follow.

(i) Fuel heater. Usually these devices are fuel-to-engine oil heat exchangers and are normally located to protect the fuel filter from blockage by ice in the fuel. The adequacy of these devices should be established. Usually this involves generation of a heat balance between heat gained by fuel and heat lost by oil using performance data provided by the manufacturers of the fuel-oil heater, the oil cooler, the heat rejected by the engine to the oil, etc. A minimum oil temperature associated with the adequacy of the fuel heater may need to be established, marked on the oil temperature gauge, and verified to be maintained during critical flight conditions. Other unprotected parts of the fuel system remain to be evaluated and substantiated for compliance with this requirement.

(ii) Oversized fuel filter. This method may only substantiate the fuel filter and, as with the fuel heater method, is incomplete without evaluation of the remainder of the fuel system. An icing test of the filter should be accomplished. Fuel preparation procedures and method of testing should follow the applicable portion of SAE Aerospace Recommended Practice (ARP) No. 1401. A satisfactory configuration is achieved when a filter is demonstrated to have the capacity to continue to provide the filtration function, without bypassing, when subjected to fuel contaminated by ice to the degree required by this rule. Usually, a delta pressure caution signal for the filter is needed to alert the flight crew that progressive filter blockage is in progress. The caution device setting should be established by test which demonstrates that after illumination of the caution signal sufficient filter capacity exists to enable completion of the flight. Fuel pressure should not fall below established limits because of ice accumulation on the filter.

(iii) Anti-ice additives. This method utilizes the properties of ethylene glycol to reduce the freezing temperature of water in the fuel. It has the advantage over other methods of protecting all components in the fuel system from ice blockage. Compliance with the rule by this method involves the following.

(A) Eligible additives. PFA-55MB (Phillips Petroleum Co.) and additives per specification MIL-I-27868, Revision D, or earlier. Later versions of this specification do not require glycerin, which may be needed to protect fuel tank coatings.

(B) Compatibility. Both engine fuel system and aircraft fuel system should be verified to be chemically compatible with the additive at the maximum concentration to be expected in the fuel system. Usually, information on eligible system materials can be obtained from the engine manufacturer for the engine fuel system and from the additive manufacturer for aircraft fuel system materials.

(C) Adding or blending the additive to the fuel. These additives do not mix well with the fuel and indiscriminate dumping of additive into the tank will not only fail to protect the system from ice accumulation but likely will damage nonmetallic components in the system. Some fuels may have additive premixed in the fuel. If other fuels are to be eligible, a method for blending additive into the fuel during refueling must be devised and demonstrated to be effective.

(D) Placards should be added near the fuel filler opening to note that fuel must contain the anti-ice additive PFA-55MB MIL-I-27686 within the minimum and maximum allowed concentration.

(E) The FAA/AUTHORITY-approved flight manual should contain necessary information to attain satisfactory blending of the additive and procedures to allow the operator to check the blend in the fuel tank.

(iv) Fuel system protection (other than filters). If the fuel heater method or oversize filter method (paragraphs AC 27.951b(3)(i) and b(3)(ii)) is proposed, the remainder of the fuel system should be shown to be free from obstruction by fuel ice. This may be shown by testing the system with ice-contaminated fuel (prepared as suggested for filter tests) or, in many cases, by selecting fuel system components which by test or by previous experience are known to be free of ice collection tendencies. Tank outlet screens (or tank-mounted pump inlet screens) may be the significant fuel system feature for further evaluation. In some instances, fuel turbulence due to pump motions may be sufficient to keep the screen clear of ice. In other instances, small screen bypass openings (approximately one-fourth inch in diameter) located outside the predominant fuel flow path have been found satisfactory.

NOTE: Advisory Circular (AC) 20-29 contains information regarding compliance with the fuel ice protection requirements of Part 25, § 25.997(b). The information in this AC is largely valid except for references to the quantity of water to be expected in fuel and the amount of additive required to ensure freedom from fuel ice hazards.

AC 27.952. § 27.952 (Amendment 27-30) FUEL SYSTEM CRASH RESISTANCE.

a. Explanation.

(1) Section 27.952 (added by Amendment 27-30) provides safety standards that minimize postcrash fire (PCF) in a survivable impact. The rule contains comprehensive crash resistant fuel system (CRFS) design and test criteria that significantly minimize fuel leaks, creation of potential ignition sources, and the occurrence of PCF. Section 27.952 accomplishes this for survivable impacts by:

(i) Providing comprehensive criteria to minimize fuel leaks and potential ignition sources;

(ii) Requiring increased crash load factors for fuel cells in and behind occupied areas to ensure the static, ultimate strength necessary for impact energy absorption, structural integrity, fuel containment, and occupant safety;

(iii) Maintaining the load factors of § 27.561 for fuel cells in other areas (particularly underfloor cells) to ensure leak-tight fuel cell deformation in energy absorbing underfloor structure without unduly crushing or penetrating the occupiable volume; and

(iv) Requiring a 50 ft. dynamic vertical impact (drop) test to measure fuel tank structural and fuel containment integrity.

(2) Section 27.952 applies to all fuel systems (including auxiliary propulsion unit (APU) systems).

(3) Some similarities exist among the fire protection requirements of §§ 27.863, 27.1337(a)(2), and 27.952. The requirements in each standard are not mutually exclusive. Overlapping requirements should be certified simultaneously.

(4) The use of bladders is not mandated as this would unduly dictate design. However, in the majority of cases, their use is necessary to meet the test requirements of § 27.952. If a design does not use bladders, the application should be treated as a new and unusual design feature and should be thoroughly coordinated with the Airworthiness Authority for Technical Policy to insure adequate safety. Experience has shown that bladders with wall thicknesses from 0.03 to 0.018 inches typically meet the § 27.952 test requirements.

b. Related Material. Documents shown below may be obtained from The Naval Publications and Forms Center, 5801 Tabor Avenue, Philadelphia, Pennsylvania 19120-5094, ATTN: Customer Service (NPODS).

(1) Military Specification, MIL-T-27422B, Amendment 1, April 13, 1971, Tank, Fuel, Crash-resistant Aircraft.

(2) Military Standard, MIL-STD-1290 (AV), January 25, 1974, Light Fixed and Rotary Wing Aircraft Crashworthiness.

(3) Military Standard, MIL-H-83796, August 1, 1974, Hose Assembly, Rubber, Lightweight, Medium Pressure, General Specification for.

(4) Military Specification, MIL-V-27393 (USAF), July 12, 1960, Valve, Safety, Fuel Cell Fitting, Crash Resistant, General Specification for.

(5) Military Specification, MIL-H-25579 (USAF).

(6) Military Specification, MIL-H-38360.

(7) U.S. Army Publication USARTL-TR-79-22E, "Aircraft Crash Survival Design Guide, Volume V---Aircraft Postcrash Survival", dated January 1989.

NOTE: Section 4, "Postcrash Fire Protection" of Volume V of the Design Guide is the modern update to MIL-STD-1290. Section 4 contains a comprehensive design guide for military CRFS designs that may be useful for civil CRFS designs.

c. Conceptual Definitions.

(1) Survivable Impact. An impact (crash) where human tolerance acceleration limits are not exceeded in any of the principal rotorcraft axes, where the structure and structural volume surrounding occupants are sufficiently intact during and after impact to constitute a livable volume and permit survival, and where an item of mass does not become unrestrained and create an occupant hazard. "Livable volume" relates to the ability of an airframe to maintain a protective shell around occupants during a crash and to minimize threats, such as accelerations, applied to the occupiable portion of the aircraft during otherwise survivable impacts. In lieu of a more rational, approved criteria, the load factors of § 27.952(b)(1) constitute the structural human survivability accelerations limits.

(2) Postcrash Fire (PCF). A fire occurring immediately after and as a direct result of an impact. The fire is either the result of fuel released from a leaking fuel system reaching an existing or a crash-induced ignition source, a crash-induced ignition source internal to an undamaged or damaged fuel system, or a combination. PCF's have an intensity range from the minimum of a small local flame to the maximum of an instantaneous massive fire or fireball (explosion).

(3) Fuel Tank or Cell. A reservoir that contains fuel and may consist of a hard shell (of a composite, metal, or hybrid construction) with either a laced-in, snapped in, or otherwise attached semirigid or flexible rubber matrix bladder (or liner), spray-on bladder, or no bladder. The hard shell may be either the airframe (integral tank) or a separate rigid tank attached to the airframe. The device has inlets and outlets for fuel transfer and internal pressure control.

(4) Ignition Source. An ignition source that when wet with fuel or in contact with fuel vapor would cause a PCF.

(5) Major Fuel System Component. A fuel system part with enough mass, installation location hazard or a combination to be structurally considered in a crash. Structural consideration is required when crash-induced relative motion can occur between the part and its surrounding structure from inertial impact forces, airframe deformation forces, or for other reasons.

(6) Drip Fence. A physical barrier that interrupts liquid flow on the underside of a surface, such as a fuel cell, and allows it to drip nonhazardously to an external drain.

(7) Flow Diverter. A physical barrier that interrupts or diverts the flow of a liquid.

(8) Frangible Attachment or Fitting. An attachment or fitting containing a part that is designed and constructed to fail at a predetermined location and load.

(9) Deformable Attachment or Fitting. An attachment or fitting containing a part that is designed and constructed to deform at a predetermined location and load to a predetermined final configuration.

(10) Self-Sealing Breakaway Fuel Fitting. A fuel-carrying in-line, line-to-firewall, bulkhead or line-to-tank connection that breaks in half and self-seals when subjected to forces greater than or equal to the unit's design breakaway force. Each half self-seals using a spring-loaded valve (e.g., trap door or equivalent means) that is normally open but is released and closed upon fitting separation. Fitting breakaway force is typically controlled by a frangible metal ring (or series of circumferential tabs) that connects the two fitting halves. Normal, fuel-tight integrity is maintained by "O" rings held under pressure by the rigid, frangible connecting ring (or tabs). When broken open, a small amount of fuel (usually less than 8 ounces) is released. This is the fuel trapped in the coupling space between the two spring-loaded valves. Once failed each coupling half may leak slightly. Typically, this leak rate should be less than 5 drops per minute per coupling half.

(11) Crash Resistant Flexible Fuel Cell Bladder. Flexible, rubberized material, usually with fibers (i.e., rubber "resin" and natural or synthetic fiber) in both the 0° (warp) and 90° (fill) directions that is used as a liner in a rigid shell or integral tank. The material acts as a membrane because, when unsupported, it can only carry pure tension loads. Therefore, it must be uniformly supported by rigid structure (reference § 27.967) so that the liner carries only compressive fluid loads and the surrounding shell structure carries the fluid-induced shear, tension, and bending loads transmitted through the liner or bladder. The material is usually secured (e.g., laced, snapped, etc.) into its surrounding structure at key locations to maintain its intended conformal shape. In many designs, lightweight spacers, such as structural foam, are used between the liner and the airframe to maintain the liners intended conformal shape and to transmit fluid loads to the airframe. The material is either qualified under TSO-C80, "Flexible Fuel and Oil Cell Material," or qualified during certification. Sections 27.952 and 27.963(g) have increased the minimum puncture resistance qualification requirement for liner material (see TSO-C80, paragraph 16.0) from 15 to 370 pounds.

(12) Crash Resistant Fuel System (CRFS). A fuel system designed and approved in accordance with § 27.952 that either prevents a PCF or delays the start of a severe PCF long enough to allow escape.

(13) As Far as Practicable. "As Far as Practicable" means that within the major constraints of the applicant's design (e.g., aerodynamic shape, space, volume, major structural relocation, etc.), this standard's criteria should be met. The level of practicability is much higher in a new design project than in a modification project. The engineering decisions, evaluations, and trade studies that determine the maximum level of practicability should be documented and approved.

(14) Fireproof. Defined in § 1.1, "General Definitions" and in AC 20-135, "Powerplant Installation and Propulsion System Component Fire Protection Test Methods, Standards and Criteria" dated February 6, 1990.

d. Procedures.

(1) Section 27.952 should be applied to all fuel system installations. Any major design change should be reevaluated for compliance with the CRFS requirements. It should be noted that most standard materials and processes are acceptable for crash resistant fuel system construction; however, magnesium, magnesium alloys, and cadmium plated parts (when exposed to fuel) are not recommended, because of their inherent ability to create or contribute to a post crash fire. Section 27.952(a) requires each tank, or the most critical tank (if clearly identified by rational analysis) to be drop tested. The tank is filled 80 percent with water and the remaining 20 percent is filled with air (or, in the case of a flexible fuel cell, the air may be evacuated by hand and the cell resealed). The tank openings, except for the vents, are closed with plugs (or other suitable means) so that they remain watertight. The vents are left open to simulate natural venting. Otherwise, the tank is flight configured. The test tanks are installed in their surrounding structure and dropped from a height of 50 feet on a nondeformable surface (e.g., concrete or equivalent). To be considered a valid test, the tank must impact horizontally ±10°. The 50-foot distance is measured between the nondeformable surface and the bottom of the tank. The ±10° attitude requirement can be ensured by using lightweight cord or a light sling to balance the tank assembly horizontally prior to being dropped. MIL-T-27422B shows a typical test setup. Tank attitude at impact should be verified by photography or equivalent means. The nondeformable floor surface should be covered by a thin plastic sheet so that any leakage is readily detected. The tank water should be tinted with dye to make leakage and seepage sources easy to identify. The tank (except for the vent openings) should be wrapped in light plastic sheet to ensure that minor leakage or seepage (and its source) is detected. Minor spillage through the open vents during the drop test is allowed. The dye should not significantly affect the water's viscosity or other physical properties that may reduce or eliminate any leakage from the drop test. The nondeforming drop test surface should be carefully reviewed. Concrete is acceptable. A fixed and uniformly supported steel plate (loaded only in uniform compression without any springback) is acceptable. Floors or floor coverings such as dirt, clay, wood, or sand are not acceptable. Selection

of the critical fuel tank is important. Factors such as size, fuel cell design and construction, and material(s) should be accounted for when selecting the critical tank. The applicant may elect to drop only a bare fuel cell, not a surrounding structural airframe segment with a fuel cell installed. If so, the applicant must show that puncture hazards to the fuel cell have been eliminated.

(i) If the applicant elects to perform the drop test with surrounding aircraft structure, the cell should be enclosed in enough surrounding structure (production or simulated) so that the airframe/fuel tank interaction during the 50-foot drop is realistically evaluated. This allows the fuel-tight integrity of the “as installed” fuel cell to be evaluated and may provide protection in some designs due to the energy absorption of the surrounding airframe when crushed by impact. This provides realistic testing of fuel cell rupture points caused by installation design features, projections, excessive deformation and local tearout of fittings, joints, or lacings. The amount of actual (or simulated) structure included in the test requires engineering evaluation, risk assessment, and detailed analysis and may require subassembly (e.g., joint) tests for proper determination. Typically, the structure surrounding and extending 1 foot forward and aft of the fuel cell is adequate. This structure has a high probability of causing crash-induced fuel cell leakage. Each application should be examined individually to include all potential structural hazards. If the surrounding structure is clearly shown not to be a contributing hazard for the drop test, and if the applicant elects to do so, the fuel cell may be conservatively dropped alone. This determination should be carefully made by a detailed engineering evaluation. The evaluation should use standard, finite element-based programs (e.g., ‘KRASH”, NASTRAN, etc.) or similar programs submitted during certification, subassembly or component tests. Elimination of the surrounding structure for the drop test configuration is not trivial. If elimination is applied for, the data should clearly and conclusively show that the surrounding structure is not an impact hazard. In any case, the drop height is a constant 50 feet. The work that determines the test article configuration should be summarized, documented, and approved.

(ii) If the drop test is used to show partial compliance with the underfloor fuel cell load factors of § 27.952(b)(3), test plans should be approved. Minor spillage from the open vents is allowed. Full compliance to these load factors should be shown by static analysis and/or tests. The intent is to provide a fuel cell that is fuel tight and does not unduly crush the occupiable volume or overly stiffen energy absorbing underfloor structure under vertical impact.

(iii) Immediately after the drop test, the tank should be placed in the same axial orientation from which it was dropped and visually examined for leakage. Minor spillage from the open vents is allowed. After 15 minutes, the tank should be reexamined and any new leakage or seepage sources noted and recorded. Any evidence of fluid on the plastic floor cover or tank wrapping sheet should be noted and recorded. Any fluid leakage or seepage constitutes a test failure. This procedure should be repeated immediately with the tank inverted and the vents plugged. The inversion procedure will identify any leak sources on the upper surfaces.

(2) Section 27.952(b) provides three sets of static load factors for design and static analysis of fuel tanks, other fuel system components of significant mass and their installations. "Installation" is structurally defined as the fuel cell's attachment to the airframe and any additional local (point design) airframe structure affected significantly by fuel cell crash loads (i.e., that would fail or deform to the extent that a fuel spill or a ballistic hazard would occur in a survivable impact). Section 27.952(d) significantly limits the amount of local airframe structure to be considered. The provision of load factors by zone ensures the fuel-tight integrity necessary to minimize PCF in a survivable impact. Unless explicitly shown by both analysis and test that the probability of fuel leakage in a survivable impact is 1×10^{-9} or less, each tank and its installation must be designed and analyzed to one set of these load factors. Also, as stated and explained in the advisory material for § 27.561, the load factors specified by § 27.561(d) are for the airframe structure surrounding the fuel cell only. The fuel cells themselves (and any fuel system components of significant mass in the underfloor area) and their attachments to the surrounding airframe structure are subject to the load factors of § 27.952(b)(3).

(i) Section 27.952(b)(1) provides load factors for the design and static analysis of fuel cells and their attachments inside the cabin volume. These load factors are provided to prevent crash-induced fuel cell ballistics hazards to and fuel spills (that may cause a PCF) directly on occupants from local structural failures in a survivable impact.

(ii) Section 27.952(b)(2) provides load factors for design and static analysis of fuel cells and their attachments located above or behind the cabin volume. These load factors are provided to prevent injury or death from a fuel cell behind or above the occupied volume that is loosened by impact and to prevent fuel spills (which may cause a PCF) in a survivable impact.

(iii) Section 27.952(b)(3) provides load factors identical to those of § 27.561 for design and static analysis of fuel cells and attachments located in areas other than inside, behind, or above the cabin volume. Since many fuel cells are located under the cabin floor, these load factors provide fuel-tight structural protection in a survivable impact.

(iv) For some crash resistant semi-rigid bladder and flexible liner fuel cell installations, the 50-foot drop test (reference § 27.952(a)) can (with some additional rational analysis) simultaneously satisfy both the drop test requirement and the vertical down load factor ($-N_Z$) requirement of § 27.952(b)(3) for the fuel cell itself and its installation. This approach reduces the certification burden.

(v) For applicants that seek to substantiate the $-N_Z$ load factor requirement of § 27.952(b)(3) using the 50-foot drop test, additional substantiation is required for § 27.952(b)(3) (as is currently practiced) for the fuel cell under the loading of the remaining three load factors and the remaining rotorcraft structure under the loading of all four load factors. In some cases substantiation of the remaining three load factors can be further simplified by a successful drop test if the fuel cell is symmetric (i.e., structurally equivalent in all four directions).

(3) Section 27.952(c) requires self-sealing breakaway fuel fittings at all fuel tank-to-line connections, tank-to-tank interconnects, and other points (e.g., fuel lines penetrating firewalls or bulkheads) where a reasonable probability (as determined by engineering evaluation, service history, analysis, test or a combination) of impact-induced hazardous relative motion exists that may cause fuel leakage to an ignition source and create a PCF during a survivable impact. In some coupling installations (such as fuel line-to-fuel tank connections), the tank coupling half should be sufficiently recessed into the tank or otherwise protected so that hazardous relative motion (of the fuel cell relative to its surroundings) following an impact-induced coupling failure does not cause a tearout or deformation of the tank half of the separated coupling that would release fuel. The only exceptions are either-

(i) Installations that use equivalent devices such as extensible lines (hoses with enough slack or stretch to absorb relative motion without leakage) or motion absorbing fittings (rotational or linearly extensible joints); or

(ii) Installations that conclusively show by a combination of experience, tests, and analysis to have a probability of fuel loss to an ignition source in a survivable crash of 1×10^{-9} or less.

(4) Section 27.952(c)(1) specifies the basic design features required for self-sealing breakaway couplings.

(5) Section 27.952(c)(1)(i) defines the design load (strength) conditions necessary to separate a breakaway coupling. These loads should be determined from analysis and/or test, reference paragraph d(6). The minimum ultimate failure load (strength) is the load that fails the weakest component in a fluid-carrying line based on that component's ultimate strength. This load comes from local deformation between the coupling and its surrounding structure during a worst-case survivable impact. A failure test of three specimens of the weakest component in each line that contains a coupling should be conducted in the critical loading mode. (If a single critical loading mode cannot be clearly identified, each of the three most critical loading modes should be tested.) The three specimen test results should be averaged. The average value is then used to size the breakaway fuel coupling. [For standard specification (i.e., "off the shelf") hardware, equivalent testing may have already been accomplished and, if no other mitigating circumstances in the design and installation exist, need not be repeated.] To assure separation of the coupling prior to full line failure and to prevent inadvertent actuation, the design load that separates the coupling should be between 25 and 50 percent of the minimum ultimate failure load (strength) of the line's weakest component. The critical loads should be compared to the normal service loads calculated and measured at the coupling location to insure unintended service failures do not occur. Typically this criterion is readily satisfied by the natural design because

working loads are much less than crash-induced loads. A separation load less than 300 pounds should not be used regardless of the line size. The minimum 300-pound load is necessary to prevent ground maintenance failures. A fatigue analysis and/or test (reference paragraph d(10)) should be performed to ensure the installation is either a safe-life design or has a conservative, mandatory replacement time. The simplified method of Section 9a of AC 20-95 may normally be used because of the low ratio of working load-to-crash-induced failure load. However, since fatigue failures have occurred in service, all fatigue sources (especially high-cycle vibratory sources) should be evaluated. Fracture critical materials should be avoided, and damage tolerant materials utilized. Also, if airframe deformation due to flight loads is significant, its effect on the couplings should be checked to ensure that static or low-cycle fatigue failures do not occur prior to the part's intended retirement life. Large flight load deformations are not usually present in rotorcraft.

(6) Section 27.952(c)(1)(ii) requires a self-sealing breakaway coupling to separate when the minimum breakaway load (reference paragraph AC 27.952d(5) and § 27.952(c)(1)(i)) is met or exceeded in a survivable impact. The loading modes (each of which produces a breakaway load) are determined by analyzing and/or testing the surrounding structure to determine the probable impact forces and directions. The modes usually occurring are tension, bending, shear, compression, or a combination (figure AC 27.952-1). The coupling should be designed and tested to separate at the lowest ultimate impact load (lowest critical mode) as long as the minimum working load criterion of § 27.952(c)(1)(i) is also satisfied. Each breakaway coupling design should be tested in accordance with the following (reference MIL-STD-1290) or equivalent procedures. It should be noted that the ratio of the ultimate failure load of the weakest component in the fuel line and the normal service load (i.e., the peak load or approved clipped peak load experienced during a typical flight) of that component should be as high as possible and still meet the other load criteria of this section. Typically, this ratio should not be less than 5.

(i) Static Tests. Each breakaway coupling design should be subjected to tension and shear loads to verify and establish the design load required for separation, nature of separation, leakage during valve actuation, general valve functioning, and leakage following valve actuation. The rate of load application should not be greater than 20 inches per minute. Tests to be used where applicable are shown in figure AC 27.952-1.

(ii) Dynamic Tests. Each breakaway coupling design should be proof-tested under dynamic loading conditions. The couplings should be tested in the three most likely anticipated modes of separation as defined in paragraph d(5). The test configurations should be similar to those shown in figure AC 27.952-1. The load should be applied in less than 0.005 second, and the velocity change experienced by the loading jig should be 36 ±3 feet per second.

(7) Section 27.952(c)(1)(iii) requires that breakaway couplings be visually inspectable to determine that the coupling is locked together (fuel-tight) and remains

open during normal operations. Visual means (such as, an axial misalignment between the two coupling halves, a designed-in visual indicator, a combination or other acceptable criteria) should be considered and specified in the maintenance manual rejection criteria for operational inspections. Inspectability and phased inspection requirements should be evaluated. Special inspections after severe maneuvers or hard landings should be required.

(8) Section 27.952(c)(1)(iv) requires breakaway couplings to have design provisions that prevent uncoupling or unintended closing by operational shocks, vibrations, or accelerations. These provisions depend on both the coupling's design and installation location. The structural environment should be defined, analyzed, and compared with coupling specifications and certification data so that inadvertent decoupling or closing does not occur. A phased inspection requirement should be considered.

(9) Section 27.952(c)(1)(v) requires a coupling design to not release more than its entrapped fuel quantity when the coupling has separated and each end is sealed off. The entrapped fuel is determined by the coupling design and is essentially the fuel trapped between the seals when separation occurs (see breakaway coupling definition). This is usually less than 8 ounces of fuel per coupling. Most coupling designs will leak slightly after separation. This is acceptable but the leak rate should be 5 drops per minute, or less, per coupling half. Specifications defining the entrapped volume of fuel should be approved. If the coupling is not approved or manufactured to an acceptable military or civil specification, the qualification testing of d(6) should be conducted.

(10) Section 27.952(c)(2) requires that each breakaway coupling or equivalent device either in a single fuel feed line or a complex fuel feed system (e.g. a multiple feed line or multitank cross feed system) be designed, tested, installed, inspected, maintained, or a combination, so that the probability of inadvertent fuel shutoff in flight is 1×10^{-5}, or less, as required by § 27.955(a). This should be determined by reliability and failure analysis, other analysis, tests, or a combination and should be documented and approved. Continued airworthiness should be ensured by phased inspections, specific component replacement schedules, or a combination. This section also requires each coupling or equivalent device to meet the fatigue requirements of § 27.571 to prevent leakage. (See the fatigue discussion in paragraph d(5).) The typical method of compliance with § 27.571 used for rotor system parts may not be necessary to meet § 27.952(c)(2). An S-N curve may not need to be generated using full-scale specimen fatigue tests if the conservative method of Section 9(a) of AC 20-95, "Fatigue Evaluation of Rotorcraft Structure" can be applied successfully.

(11) Section 27.952(c)(3) requires that an equivalent device, used instead of a breakaway coupling, not produce a load, during or after a survivable impact, on the fuel line to which it attaches greater than 25-50 percent of the ultimate load (strength) of the line's weakest component. This minimizes crash-induced fuel spills that may cause a PCF. The ultimate strength of the weakest component should be determined by analysis and/or tests. At least three specimens of the component should be tested to

failure in the critical loading mode and the results averaged. [For standard specification (i.e., "off the shelf") hardware, equivalent testing may have already been accomplished and, if no other mitigating circumstances in the design and installation exist, need not be repeated.] The average value is then used to size the equivalent device. Each equivalent device must meet the fatigue requirements of § 27.571 to prevent fatigue-induced leakage. Equivalent devices should be statically and dynamically tested in an identical manner (where feasible) to breakaway couplings (reference paragraph d(6)). All fuel hoses and hose assemblies (whether or not they are used in lieu of breakaway fittings) should meet the following (reference MIL-STD-1290) or equivalent requirements. Any stretchable hoses used as equivalent devices should be able to elongate a minimum of 20 percent without leaking fuel. All other hoses used as equivalent devices should have a minimum of 20-30 percent slack. It should be noted that the ratio of the ultimate failure load of the weakest component in the fuel line and the normal service load (i.e., the peak or approved clipped peak load experienced during a typical flight) of that component should be as high as possible and still meet the other load criteria of this section. Typically, this ratio should not be less than 5.

(i) All hose assemblies should meet or exceed the cut resistance, tensile strength, and hose-fitting pullout strength criteria of MIL-H-25579 (USAF), MIL-H-38360, or equivalent standards.

(ii) Hoses should neither pull out of their end fittings nor should the end fittings break at less than the minimum loads shown in figure AC 27.952-3 when the assemblies are tested as described in d(11)(iii) below. In addition to the strength requirements, the hose assemblies should be capable of elongating to a minimum of 20 to 30 percent by stretch, slack, or a combination without fluid spillage.

(iii) Hose assemblies should be subjected to pure tension loads and to loads applied at a 90° angle to the longitudinal axis of the end fitting, as shown in figure AC 27.952-2. Loads should be applied at a constant rate not exceeding 20 inches per minute.

(12) Section 27.952(d) requires frangible or deformable structural attachments to be used to install fuel tanks and other major system components to each other and to the airframe when crash-induced hazardous relative motion could cause local rupture and tearout of the component, spill fuel to an ignition source, and create a PCF. If it can be conclusively determined that the probability of fuel spillage is 1×10^{-9} or less, no further action is required. Typically, frangible designs are much easier to certify than deformable designs because the scatter in failure loads is much less. Also, some standard frangible military hardware (e.g., frangible bolts) is readily available. This is not so for deformable designs. Each frangible or deformable structural attachment and its installation should be reviewed to insure that, after an impact failure (i.e., separation or deformation), it does not become a puncture or tear-out hazard and cause fuel spillage.

(13) Section 27.952(d)(1) defines the impact design load conditions necessary to deform a deformable attachment or to separate a frangible attachment. These loads should be determined from analysis and/or test (reference paragraph d(14)), and verified during certification. All impact loading modes (tension, bending, compression, shear, and a combination) should be analyzed and the minimum critical frangible or deformable design load determined, based on the ultimate strength of the attachment's weakest component. The critical load should be compared to the normal service loads calculated and measured at the attachment's location to insure unintended service failures do not occur. (Normally, this criterion is readily satisfied because working loads are much less than impact loads.) A fatigue check should be conducted to ensure that the attachments meet the requirements of § 27.571. Typically, this can be accomplished using the simplified method of Section 9(a) of AC 20-95 because of the low ratio of working-load-to-crash-induced failure load. However, because of service history, all fatigue sources (especially high cycle vibratory sources) should be reviewed. The standard method of compliance with § 27.571 used for rotor system parts may not be necessary to meet § 27.952(d)(3). An S-N curve may not need to be generated using full-scale specimen fatigue tests, if the conservative method of Section 9(a) of AC 20-95 can be applied successfully. Fracture critical materials should be avoided and ductile, damage tolerant materials utilized. Phased inspections to ensure continued airworthiness should be considered. Special inspections after severe maneuvers or hard landings should be required. A breakaway or deformation load less than 300 pounds (based on maintenance considerations) is not permitted. If airframe deformation due to flight loads is significant, its effect should be checked to ensure that a static failure or low cycle fatigue failure does not occur. Large flight load deflections are not usually present in rotorcraft.

(14) Section 27.952(d)(2) requires a frangible or locally deformable attachment to function when the minimum breakaway or deformation load (reference § 27.952(d)(1)) is met or exceeded in a survivable impact. The minimum breakaway or deformation load is the load that either breaks or deforms each of the frangible or deformable attachment(s) of each fuel cell, fuel line, or other critical fuel system component to the airframe. Each breakaway/deformation load must be between 25 percent to 50 percent of the load which would cause failure (i.e., impact induced tearout and subsequent fuel leakage) of the attachment to fuel cell, fuel line, or other critical component interface. This is necessary in some installations to prevent tearout of the structural attachment from the fuel cell component to which it is attached and the resultant fuel leakage in a survivable impact. The primary loading modes (each of which will produce a breakaway or deformation load) must all be considered to determine the minimum load. This is done by analyzing the surrounding structure (reference paragraph d(13)) to determine the three most probable impact failure forces and their directions. The attachment should then be tested to insure it breaks or deforms at the lowest ultimate crash (impact) load as long as the minimum working load criterion of § 27.952(d)(1) is also satisfied. It should be noted that the ratio of the ultimate failure load of the weakest component in the frangible or deformable component's load path and the normal service load (i.e., peak load or approved clipped peak load experienced during a typical flight) of that component should be as high as

possible and still meet the other load criteria of this section. Typically this ratio should not be less than 5. The following certification tests (reference MIL-STD-1290) or equivalent should be conducted on each frangible or deformable attachment design.

(i) Static Tests. Each frangible or deformable device should be tested in the three most likely anticipated modes of failure as defined in paragraph d(13). Test loads should be applied at a constant rate not exceeding 20 inches per minute until failure occurs.

(ii) Dynamic Tests. Each frangible or deformable attachment should be tested under dynamic loading conditions. The attachment should be tested in the three most likely failure modes as determined in paragraph d(13). The test load should be applied in less than 0.005 second, and the velocity change experienced by the loading jig should be 36 ±3 feet per second. It should be noted that the dynamic load pulse is a ramp function starting at either previously determined failure load in 0.005 seconds. The velocity change of the test jig 0 or some small test fixture preload and reaching the is also a ramp function starting at 0 and reaching a final velocity of 36±3 ft./sec. in 0.005 seconds. These ramps functions simulate the dynamic conditions of a survivable impact under which the frangible/deformable attachment must perform its intended function.

(15) Section 27.952(d)(3) requires a frangible or locally deformable attachment to meet the fatigue requirements of § 27.571 to eliminate premature fatigue failure. The simplified method of AC 20-95 may be used. Because of service history, all fatigue sources (especially high cycle vibratory sources) should be reviewed. Fracture critical materials should be avoided and ductile, damage tolerant materials utilized.

(16) Section 27.952(e) requires that, as far as practicable, fuel and fuel containment devices be adequately separated from occupiable areas and potential ignition sources. Several generic categories of ignition sources and potential PCF-producing contact scenarios exist. The intent of the section is to define all possible leak and ignition sources that could be activated in a survivable impact and to provide design features to eliminate or minimize them such that the occurrence of PCF is minimized and escape time is maximized. Adequate separation should be accomplished by a thorough design review, potential PCF hazard analysis, and detailed design trade studies. The resultant findings should be documented and approved. The following PCF hazards and any other such hazards should be documented, minimized by design to the maximum practicable extent, and their resolution documented and FAA/AUTHORITY approved. Conditions to be reviewed should include, but are not limited to, the following:

(i) High temperature ignition sources.

(A) Tank fillers or overboard fuel drains should not be located adjacent to engine intakes or exhausts so that fuel vapors could be ingested and ignited.

(B) Fuel lines should not be located in any occupiable area unless they are shrouded or otherwise designed to prevent spillage and subsequent ignition during and immediately following a survivable impact.

(C) Fuel tanks should not be located in or immediately adjacent to engine compartments, engine induction or exhaust areas, heaters, bleed air ducts, hot air-conditioning ducts, or any other hot surface.

(D) Fuel lines should be kept to a minimum in the engine compartment. Fluid lines should not be located immediately adjacent to engine exhaust areas, heaters, bleed air ducts, hot air-conditioning ducts, or any other hot surface.

(E) Fuel lines should not be located where they can readily spill, spray, or mist onto hot surfaces or into engine induction or exhaust areas. These locations should be determined for each aircraft design by considering probable structural deformation hazards in relation to the fuel system.

(ii) <u>Electrical ignition sources</u>.

(A) Fuel tanks and lines should not be located in electrical compartments.

(B) Electrical components and wiring should be separated from fuel lines and vent openings and kept to a minimum in fuel areas.

(C) Electrical wiring should be hermetically sealed and equipment should be explosion-proofed in areas where they are immersed in or otherwise directly subjected to fuel and vapors and should meet § 27.1309 or should otherwise be protected such that ignition is extremely improbable.

(D) Electrical sensor lines that penetrate fuel tank walls should be protected from abrasion or guillotine cutting during a survivable impact by use of potting, rubber plugs or grommets, or other equivalent means and should be designed with sufficient local slack, or equivalent means, to prevent both the wires and their protective mountings from being cut by or torn from fuel tank walls by local deformation.

(E) Electrical wires should be designed with sufficient slack or equivalent means to accommodate structural deformation without creating an ignition source.

(F) Electrical wires that could be subjected to severe local abrasion, cutting, or other damage during a survivable impact should be protected locally by nonconductive shields or shrouds.

(G) Electrical wires that are not sufficiently separated from heat or ignition sources to avoid potential contact during a survivable impact should be locally shrouded with a nonconductive fireproof shroud.

(iii) Friction spark, chemical, and electrostatic ignition sources. Fuel lines and tanks should be designed and located to eliminate fuel or fuel vapor ignition from potential mechanical friction spark ignition sources, chemical ignition sources, and electrostatic ignition sources having a high probability of being activated or created during a survivable impact.

(iv) Separation of fuel tanks and occupiable areas. Fuel tanks should be located as far as practicable from all occupiable areas. This minimizes potential PCF sources in occupiable areas and the potential for occupant saturation with fuel on impact. The design should be reviewed to minimize these potential hazards. Fuel tanks should also be removed, as far as practicable, from other potentially hazardous areas such as engine compartments, electrical compartments, under heavy masses (e.g., transmissions, engines, etc.), over landing gear, and other probable areas of significant impact damage, including rollover and skidding damage.

(v) Fuel Line Shielding. Areas of the fuel line system where the probability of spilled fuel reaching potential ignition sources or occupiable areas is greater than extremely improbable should be shielded with drainable fireproof shrouds. Shrouds should be drainable to allow periodic inspections for internal fuel leaks. The design should be reviewed to ensure these criteria are met.

(vi) Flow Diverters and Drain Holes.

(A) Drainage holes should be located in all fuel tank compartments to prevent the accumulation of spilled fuel within the aircraft. Holes should be large enough to prevent clogging by typical debris and to prevent fluid accumulation from surface tension force blockage.

(B) Drip fences and drainage troughs should be used to prevent gravity-induced flow of spilled fuels from reaching any ignition sources such as hot engine areas, electrical compartments, or other potential hot spots. Drip fences and troughs are also necessary to prevent PCF by routing spilled fuel around ignition sources to drainage holes to minimize fuel accumulation inside the fuselage. Recurring inspection requirements to ensure holes and troughs remain airworthy should be identified. These criteria should be met, as far as practicable, for all postcrash attitudes. This is readily accomplished for the standard landing attitude, but is more difficult for other abnormal attitudes. However, the design should be thoroughly reviewed to insure maximum compliance without adversely impacting other safety and design criteria such as aerodynamic smoothness.

(vii) Fuel Drain System. The fuel drain system and its attachments to the airframe should be designed and constructed, as far as practicable, to be crash resistant. The following and other appropriate means should be considered for a crash resistant design. Tank drains should be recessed or otherwise protected so that they are minimally damaged by impact. Attachment of fuel drains to the airframe should be made with either frangible fasteners or equivalent means to prevent impact induced tearout and leakage. The number of drains should be minimized by design techniques such as those that avoid low points in the lines. Drain lines should be made of ductile materials or otherwise designed to provide impact tolerance. Drain line connections, fittings, and other components should be designed to meet the fatigue requirements of § 27.571 and 27.952(d)(3). This ensures that unintended partial or full fatigue failures do not occur in normal operations that, if undetected, could compromise the CRFS's intended level-of-safety for the mitigation of post crash fire in a survivable impact. Drain valves should be designed to have positive locking provisions in the closed position in accordance with § 27.999(b)(2).

(17) Section 27.952(f) specifies that fuel tanks, fuel lines, electrical wires, and electrical devices must be designed and constructed, as far as practicable, to be crash resistant. Typical mechanical design criteria necessary to minimize fuel spillage sources, ignition sources, and their mutual contact in a survivable impact (i.e., provide crash resistance) are stated by the following subparagraphs. These mechanical design criteria should be incorporated in each design to the maximum practicable extent. Compliance is accomplished and assessed by a thorough design review and potential PCF hazard analysis with findings and solutions that are documented and approved. Any additional PCF hazards that are identified should be documented, included, addressed equally, and eliminated to the maximum practicable extent. Engineering evaluation, analysis, and tests are all required to determine the maximum level of practicability.

(i) They should not initiate or contribute to a post crash fire in an otherwise survivable impact. A hazard analysis should show which components are critical in this regard and should be assessed in detail for hazard elimination purposes.

(ii) Fuel and electrical lines and components should be located away from each other, away from probable crash impact areas, and away from areas where structural deformation or large objects (such as engines or transmissions) may, by crushing or penetration, cause fuel spillage or create an electrical ignition source, or both.

(iii) Fuel and electrical lines and components should be located separately and away from areas where impact and severing by rotor blades during a survivable impact are probable.

(iv) Fuel and electrical lines and components should be in no danger of being punctured or severed during a survivable impact by locally stiff vertical understructure such as a collapsed landing gear strut.

(v) Fuel and electrical lines and components should be routed separately in areas of maximum protection, such as along heavier structural members, and away from areas where significant damage is probable.

(vi) Fuel and electrical lines and components running through hazardous areas or directly through structure, such as a bulkhead, should be locally separated and protected from over-extension, severe abrasion and guillotine cutting by frangible panels, suitable clearance, rubber grommets, braided armor shielding (which should be nonconductive for electrical lines), or other equivalent means.

(vii) Fuel lines routed directly to instruments, transducers, or other equivalent devices should be crash resistant, in accordance with § 27.1337(a)(2), to minimize leakage in case of line rupture induced during a survivable impact.

(viii) Electrical wires routed directly into electrical boxes or instruments should be designed with sufficient local slack and locally routed in the least probable damage direction and zone, or otherwise protected to minimize the probability of damage-induced arcing.

(ix) Fuel lines routed directly into fuel tanks or other fuel system components should be locally routed in the least probable damage direction and zone, or otherwise protected, to minimize the probability of damage-induced fuel leaks.

(x) Fuel pumps mounted inside fuel tanks should be rigidly attached to the fuel tank only. If the pump is airframe mounted and has structural significance, it should have a frangible or deformable attachment (reference paragraph AC 27.952d(12)). Electrical boost pumps, if used, should be installed with a minimum of 6 inches of slack wire at the pump connection. The pump wires should be shrouded to prevent cutting in a survivable impact. Nonsparking, breakaway wire disconnects or other equivalent means may be used in lieu of the 6 inches of slack wire.

(xi) Fuel filters and strainers, to the maximum practicable extent, should not be located in or adjacent to the engine intake or exhausts and should retain the smallest practicable quantity of fuel.

(xii) The number of fuel valves should be kept to a minimum. If electrically operated valves are used, they should be installed with a minimum of 6 inches of slack in the electrical lines, unless protected by equivalent mean (reference 17(i)). The valves should be installed with the maximum amount of protection and separation of the electrical wires from the remainder of the valve assembly.

(xiii) Fuel quantity indicators mounted in or on fuel tanks should be selected, designed, and installed to provide the minimum puncture or tear hazard to the fuel tank in a survivable impact.

(xiv) Fuel tank and bladder enclosures should have smooth, regular shapes that avoid sharp edges and corners. Minimum concave and convex radius design criteria should be developed and adhered to. Magnesium should not be used in fuel cells, and any cadmium-plated parts should not be exposed to fuel.

(xv) Any shielding of electrical wires from abrasion, cutting, or overextension must be nonconductive.

(xvi) All fuel line installations not containing breakaway couplings should be reviewed to insure that they will not be overtensioned in a survivable impact, that they are properly grouped and properly exit fuel tanks, firewalls, and bulkheads in the area of least probable damage, and that their number and lengths are safely minimized.

(xvii) Crash resistance guidance for other basic components is contained in related AC paragraphs such as paragraphs AC 27.963 (§ 27.963, bladders and liners), AC 27.973 (§ 27.973, fuel tank filler connections) and AC 27.975 (§ 27.975, fuel tank vents).

(18) Section 27.952(g) requires rigid or semirigid fuel tank or bladder walls of any material construction to be both impact and tear resistant. This minimizes a PCF from impact-induced rupture and tear.

(i) A rigid tank or bladder can resist fluid pressure loads as a flat plate in bending. A semirigid tank can resist fluid pressure loads partially as a flat plate in bending and partially as a membrane in tension. Flexible liners are exempt from the requirements of § 27.952(g) since an unsupported flexible liner can resist only pure tension loads acting as a membrane (i.e., it has negligible bending strength). The rigid shell structure required by § 27.967(a)(3) that surrounds the flexible liner (membrane) carries the crash-induced impact and tear loads; whereas, the flexible liner is only significantly loaded in tension if the shell structure is penetrated by a sharp object on impact.

(ii) For metallic tanks, rigid or semirigid composite tanks (resin matrix), semirigid bladder designs (rubber matrix), metal-composite hybrid designs, and all other tank designs, impact and tear resistance should be shown by analysis and tests.

(iii) Designs using resin matrix composites should be subjected to the composite structure substantiation guidance of AC 20-107A, Composite Aircraft Structure, dated April 25, 1984, and paragraph AC 27 MG 8. Designs using rubber matrix composites are subject to the standard substantiation requirements for these devices, such as TSO-C80.

(iv) One set of crash resistance tests that constitutes an acceptable method of substantiation to the requirements of § 27.952(g) for all tank designs regardless of the materials used are those specified in Paragraphs 4.6.5.1 (Constant Rate Tear); 4.6.5.2 (Impact Penetration); 4.6.5.3 (Impact Tear); 4.6.5.4 (Panel Strength Calibration); and 4.6.5.5 (Fitting Strength) of MIL-T-27422B, "Military Specification; Tank, Fuel, Crash-Resistant Aircraft." These test requirements, or equivalent means, should be applied for and discussed early in certification. If the MIL-T-27422B tests are selected, severity differences between military combat requirements and the civil environment should be accounted for by reducing the MIL-T-27422B requirements, as follows:

(A) Constant Rate Tear. The minimum energy for complete separation should be 200 foot-pounds (reference 4.6.5.1).

(B) Impact Penetration. The drop height of a 5-pound chisel should be reduced to 8.0 feet (reference 4.6.5.2).

(C) Impact Tear. The drop height of a 5-pound chisel should be reduced to 8.0 feet and the average tear criteria should not exceed 1.0 inch (reference 4.6.5.3).

(19) Section 27.952(g) also requires that all fuel tank designs (regardless of the materials utilized and whether or not a flexible liner of any type is used) for each tank or the most critical tank be analyzed and tested to the criteria of paragraph (18)(iv) of § 27.952, or equivalent.

(20) Any type of flexible liner or bladder used in any type of fuel tank construction (integral, hard shell, etc.) must meet the strength and puncture resistance requirements of § 27.963(g). Section 27.963(g) contains the new puncture resistance requirement for flexible liners and other liner material certification requirements. Unlined, bladderless fuel tanks are also required to meet this requirement. Most unlined, rigid fuel cell designs should readily exceed the 370-pound minimum puncture force requirement because of overriding design requirements and material characteristics, such as stiffness and ductility.

NOTE: TSO-C80, “Flexible Fuel and Oil Cell Material,” is referenced in the advisory material for § 27.963(g) and contains the detailed qualification requirements for these materials. The current puncture resistance test of TSO-C80, paragraph 16.0, states that the force required to puncture the bladder material must be greater than or equal to 15 pounds (e.g., screwdriver test). Section 27.963(g) has increased the TSO paragraph 16.0 puncture force value to be greater than or equal to 370 pounds. This is for fuel cell bladder or liner material only. Oil cell material puncture force requirements are not changed.

e. Typical Examples of Loading Modes and Test Setups for CRFS Components. The following figures, which are referred to periodically in the advisory circular, show typical examples of test setups for CRFS components such as breakaway fuel fittings, hoses, hose end fittings, and hose assemblies.

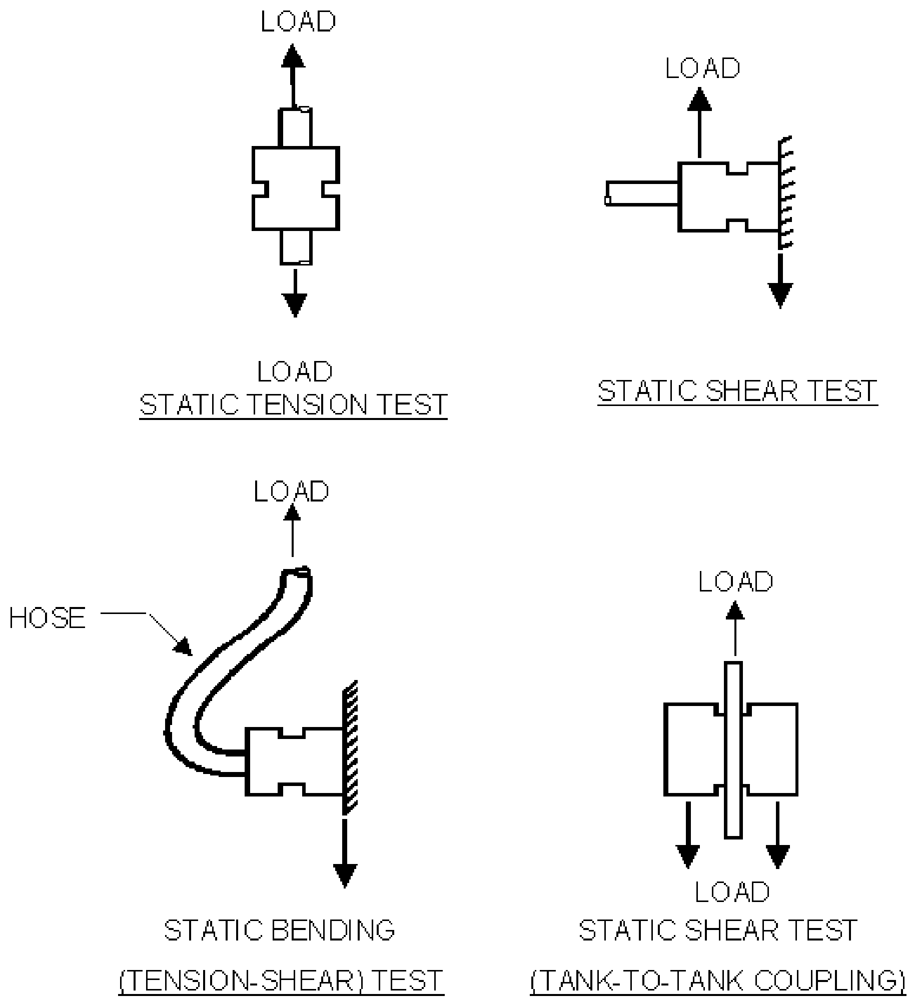
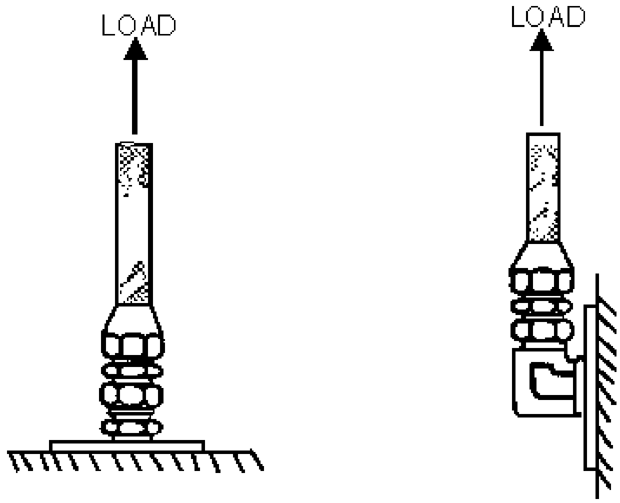
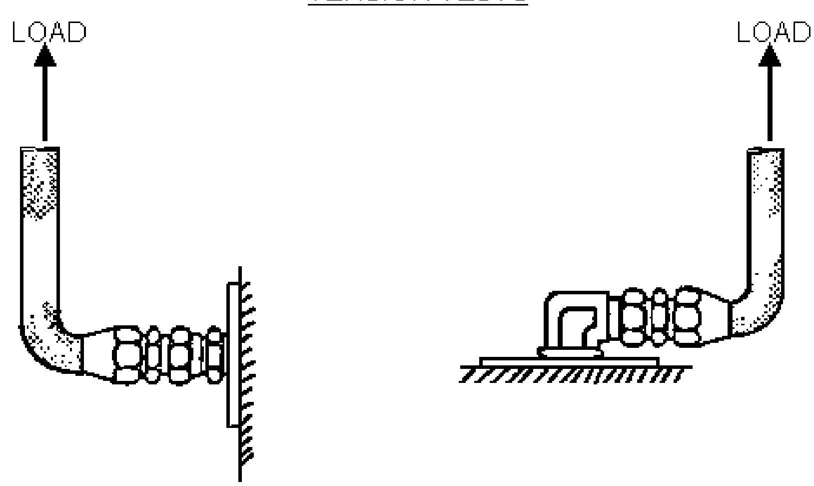


FIGURE AC 27.952-1 STATIC TENSION AND SHEAR LOADING MODES

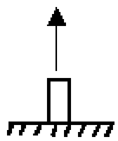
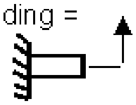
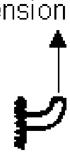
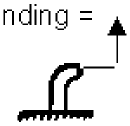


TENSION TESTS



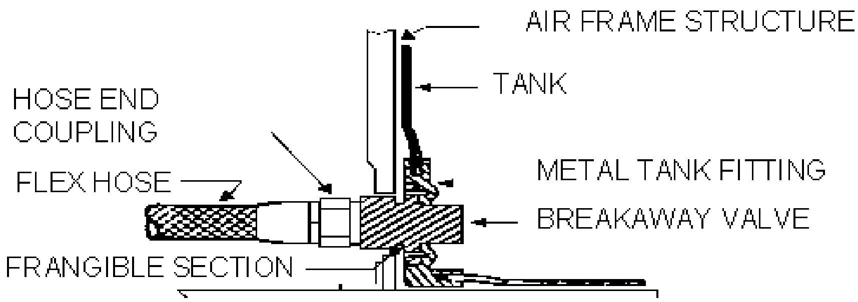
90-DEGREE TESTS

FIGURE AC 27.952-2 HOSE ASSEMBLY TESTS

Hose End Fitting Type	Fitting Size	Tension Load (lb) Minimum Average Load*	Tension Load (lb) Minimum Individual Load	Bending Load (lb) Minimum Average Load*	Bending Load (lb) Minimum Individual Load
STRAIGHT	-4	600	475	425	400
Tension =	-6	700	575	425	400
	-8	900	650	650	600
	-10	1450	1175	675	625
Bending =	-12	1775	1475	950	850
	-16	2125	1825	1425	1300
	-20	2375	2075	1550	1425
90° ELBOW	-4	600	475	425	400
Tension =	-6	700	575	425	400
	-8	900	650	450	400
	-10	1450	1175	475	425
Bending =	-12	1775	1475	500	450
	-16	2125	1825	775	700
	-20	2375	2075	1100	1000

*Average of at least 3 tests.

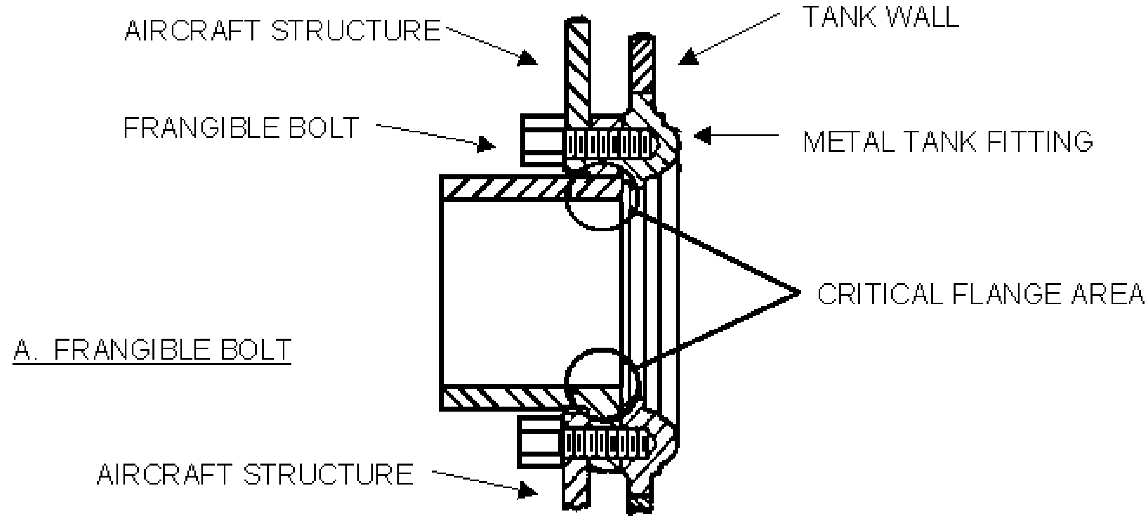
FIGURE AC 27.952-3 MINIMUM AVERAGE AND INDIVIDUAL LOADS FOR HOSE AND HOSE-END FITTING COMBINATIONS



ITEM	LOWEST FAILURE LOAD (LB)*	FAILURE MODE
Flex Hose	**3000**	**Tension Breakage**
Flex Hose	**1500**	**Pull Out of End Fitting**
Tank Fitting	**7500**	**Pull Out of Tank**
Hose End Coupling	**1650**	**Break (Bending)**
Breakaway Valve	**2500**	**Pull Out of Tank Fitting**
Breakaway Valve	**Not More Than** $\frac{1500}{2} = 750$ **Not Less Than** $\frac{1500}{4} = 375$	**Break at frangible Section**

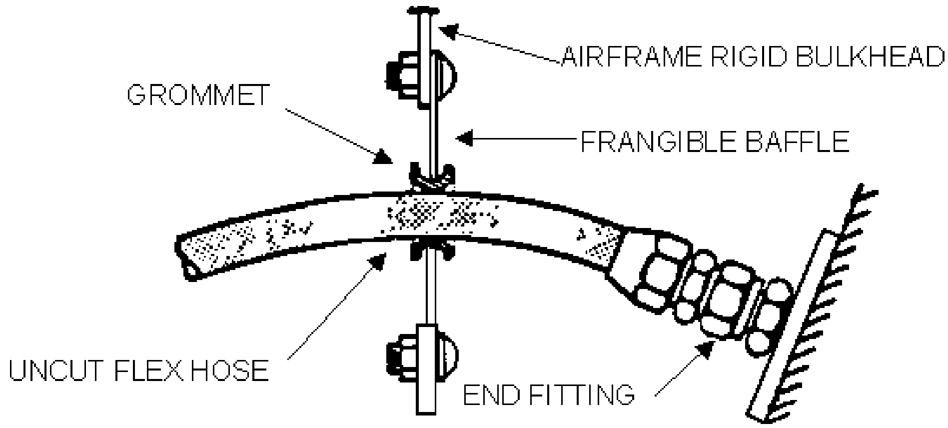
*Loads may or may not be representative; values are for explanatory purposes only.

FIGURE AC 27.952-4 TYPICAL METHOD OF BREAKAWAY FUEL FITTING LOAD CALCULATIONS (TANK INSTALLATION USED AS EXAMPLE ONLY; BASIC TECHNIQUE APPLICABLE TO OTHER CONFIGURATIONS)



ITEM	LOWEST FAILURE LOAD (LB)*		FAILURE MODE
AIRCRAFT STRUCTURE	4000		SHEAR
TANK FITTING	3000		PULLOUT OF TANK
FLANGE	5000		SHEAR
FRANGIBLE BOLT	NOT MORE THAN $\frac{3000}{2} = 1500$	NOT LESS THAN $\frac{3000}{4} = 750$	BREAK (TENSION-SHEAR)

FIGURE AC 27.952-5 TYPICAL METHODS OF FRANGIBLE OR DEFORMABLE ATTACHMENT LOAD CALCULATIONS: EXAMPLE 1, FRANGIBLE BOLTS.



ITEM	LOWEST FAILURE LOAD (LB)*		FAILURE MODE
RIGID BULKHEAD	4000		BEARING
FLEX HOSE	3000		TENSION BREAKAGE
FLEX HOSE	1500		PULLOUT OF END FITTING
END FITTING	1750		BENDING
FRANGIBLE BAFFLE	NOT MORE THAN $\frac{1500}{2} = 750$	NOT LESS THAN $\frac{1500}{4} = 375$	BEARING

*VALUES ARE SHOWN FOR EXPLANATORY PURPOSES ONLY

FIGURE AC 27.952-6 TYPICAL METHODS OF FRANGIBLE OR DEFORMABLE ATTACHMENT LOAD CALCULATIONS: EXAMPLE 2, FRANGIBLE BAFFLE.

AC 27.953. <u>§ 27.953 FUEL SYSTEM INDEPENDENCE</u>.

a. <u>Explanation</u>.

(1) Section 27.953(a) specifies independent fuel feed systems for each engine of multiengine rotorcraft; however, separate fuel tanks for each engine are not required.

(2) If a single tank is used to feed more than one engine, § 27.953(b) specifies:

(i) That independent fuel tank outlets be provided to each engine, each having a shutoff valve.

(ii) At least two vents for the tank located to minimize the probability of both vents becoming obstructed simultaneously.

(iii) Filler caps designed to minimize the probability of incorrect installation or in-flight loss.

(iv) That fuel supply from each tank outlet to any engine be independent of fuel supply to other engines.

b. <u>Procedures</u>.

(1) The purpose of § 27.953(a) is to ensure an independent fuel supply system for each engine on multiengine rotorcraft. Unlike the corresponding regulation for Category A, Part 29 rotorcraft, separate fuel tanks are not required.

(2) The assessment of an independent fuel supply system for each engine would begin at the fuel supply pickup point within the tank and continue to the engine fuel inlet at the engine.

(3) If supply line crossfeed capability is included as a feature, care must be exercised to ensure that the opening of the crossfeed does not jeopardize the continued safe operation of more than one engine. For example, if the crossfeed valve is automatically operated by a low pressure signal in the supply line for one engine, the possibility that fuel line leakage could cause opening of the crossfeed and jeopardize the continued safe operation of both engines should be considered. Similarly, opening the crossfeed valve with a suction lift system should not allow air into the fuel supply line of any engine.

(4) The independent fuel supply system requirement for each engine is for normal fuel system operations. Fuel system designs which allow the continued safe operation of all engines under expected fuel system component failure conditions (for example, a failed boost pump) by using common fuel flow paths under failure conditions are not prohibited.

(5) In § 27.953(b), the phrase “if a single fuel tank is used,” is intended to mean if a single fuel tank is used to feed more than one engine. This interpretation is needed in order to preclude, for example, a tri-engine design with two fuel tanks where two engines draw fuel by independent means from one tank, but only one vent is provided for that tank. This design would clearly violate the intent of § 27.953(b)(2) to assure that two vents be supplied if fuel is drawn by more than one engine from a single tank.

(6) If a single fuel tank is used to supply fuel to more than one engine:

(i) There should be independent tank outlets for each engine, each incorporating a shutoff valve at the tank. The phrase, “at the tank,” has rightfully been interpreted to allow the firewall shutoff valve, which may actually be some distance from the tank itself, to be used to show compliance with § 27.953(b)(1). Section 27.953(b)(1) specifically allows the shutoff valve, if located at the tank, to serve as the firewall shutoff valve provided the line between the valve and the engine compartment does not contain a hazardous amount of fuel that can drain into the engine compartment.

(ii) There should be at least two vents arranged to minimize the probability of both vents becoming obstructed simultaneously. Typically, the means used to prevent simultaneous obstruction is physical separation. The blockage or malfunction of any vent should not jeopardize the continued safe operation of more than one engine.

(iii) The filler cap(s) for the tank should be designed to minimize the probability of incorrect installation or in-flight loss. Usually, there should be only one way to install and lock a fuel cap; if more than one way is possible, either method should provide the positive sealing to avoid spillage. Minimizing the probability of in-flight fuel loss would include the ability to visually determine that the cap is properly installed and locked prior to flight.

(iv) Section 27.953(b)(4) simply clarifies that if a single tank is used to feed more than one engine, the provisions for independent fuel feed systems (reference § 27.953(a)) apply to the engines being fed from that tank.

AC 27.954. § 27.954 (Amendment 27-23) FUEL SYSTEM LIGHTNING PROTECTION.

a. Background. During the initial development and promulgation of the standards concerning the airworthiness of rotorcraft, it was not deemed necessary to specify design features that would protect the rotorcraft from the meteorological phenomenon of lightning. This was due, in part, to the fact that rotorcraft were primarily operated in a VFR and nonicing environment. Also, a prudent pilot avoided thunderstorms where the possibility of encountering severe weather and a lightning strike was much greater. The construction, design, and operating environment of civil rotorcraft have changed markedly within the past two decades. Many rotorcraft are now authorized to fly IFR. Additionally, many rotorcraft now use the same advanced technologies in structures and systems as do airplanes. Because of these facts the possibility of a lightning strike encounter to the rotorcraft has been greatly increased. If the fuel system of the rotorcraft has not been properly designed and constructed, a fuel vapor ignition may occur if the rotorcraft encounters a lightning strike. This occurrence generally results in a catastrophe to the rotorcraft. To prevent such a catastrophe and provide a level of safety equivalent to normal utility, acrobatic and commuter category airplanes, a specific rule for the lightning protection of normal category rotorcraft fuel systems was adopted in Amendment 27–23.

b. Explanation.

(1) This regulation requires that the rotorcraft's fuel system be designed and constructed so that an ignition of fuel vapor will not occur when the rotorcraft is involved in a lightning strike. For the purposes of this regulation the fuel system is comprised of the fuel tank with all its associated plumbing and any other areas of the rotorcraft likely to have fuel vapor present (such as sumps and drains for the tank itself). Externally mounted fuel tanks are also considered to be part of the "fuel system."

(2) Other associated installations such as electrical wiring in the fuel tanks which could provide a source of ignition due to an indirect or induced effect should also be considered.

c. Procedures.

(1) The current revision of Advisory Circular 20-53 provides guidance on an acceptable method and procedure to be utilized to demonstrate that the design and construction of the fuel system is compliant with § 27.954.

(2) FAA Report No. DOT/FAA/CT-89/22 contains additional information regarding the lightning environment. Also contained in this report are design and test techniques which provide for a design that will be adequately protected from fuel vapor ignition when the rotorcraft encounters the lightning environment. This report is available to the public by order from the National Technical Information Service, Springfield, VA 22161.

AC 27.955. § 27.955 FUEL FLOW.

a. Explanation.

(1) Section 27.955 is intended to ensure adequate fuel flow to the engine(s) at maximum power under the intended aircraft operating conditions and maneuvers.

(2) In showing adequate fuel flow, the rule provides--

(i) That the fuel be supplied within the appropriate engine fuel pressure range;

(ii) That the test be conducted with minimum fuel onboard, consistent with test safety; and

(iii) That operation with both main and emergency pumps be considered.

(3) Section 27.955(b) specifies that if an engine can be supplied with fuel from more than one tank, the fuel system must feed promptly when fuel becomes low in one tank and another tank is selected.

b. Procedures.

(1) Testing (including bench tests) has been the accepted method to show compliance with § 27.955(a). Analytical techniques may be used to adjust the system test results to various fuel conditions and flows or to account for minor modifications to a system. A purely analytical approach is not generally acceptable.

(2) Methods to adjust the test data for different fuel properties and flows should be verified by limited testing.

(3) If a suction lift system is used and hot fuel verification is involved, testing is appropriate.

(4) The proper interpretation of the phrase “100 percent of the fuel flow required under the intended operating conditions and maneuvers” may include consideration of acceleration fuel flow in addition to the steady-state fuel flow requirement.

(i) For example, if on a single-engine rotorcraft on a cold-day takeoff, engine torque is the limiting parameter, the steady-state fuel flow demand corresponding to that torque may be exceeded during engine acceleration in maneuvers.

(ii) In addition to the consideration of acceleration fuel flow, good design would include some margin to account for possible inadvertent overtorque.

(5) For multiengine rotorcraft, adequate fuel flow under OEI conditions should be assured in the critical fuel system configuration.

(i) If on a multiengine rotorcraft, it is acceptable to operate following an engine failure in more than one fuel system configuration (for example, if crossfeed is an acceptable mode) then the supplying of two engines through common components may be more critical than the OEI condition.

(ii) In verifying satisfactory fuel system operation for OEI conditions, the fact that the remaining engine may go to the gas producer speed topping limit fuel flow rather than to the steady-state OEI power value should be assessed.

(6) Adverse transient and steady-state maneuver loads should be considered since the g-loading experienced may tend to decrease the fuel inlet pressure below allowable limits.

(7) In assuring adequate fuel flow at the necessary engine inlet pressure (§ 27.955(a)(1)), both hot and cold fuel would normally be evaluated for the suction lift system, whereas cold fuel is usually more critical for the boosted pressure system.

(8) The method of specifying the fuel inlet pressure requirements varies with the engine model. Some of these include:

(i) Specification of a gage pressure as a function of altitude for suction system operation. The particular fuel and fuel temperature for demonstrating the criteria may be specified in the engine documents. Other approved fuels, fuel temperatures, and boost-pump-on operation are considered satisfactory if the demonstration with the specified fuel is successful.

(ii) Specification of a maximum allowable vapor-to-liquid ratio for hot fuel, and minimum absolute pressure as a function of altitude for cold fuels.

(iii) Specification of a fuel inlet pressure relative to the true vapor pressure of the fuel, in combination with a maximum allowable vapor-to-liquid ratio.

(iv) Specification of separate pressure limits for boost-on and suction lift operation.

(v) Specification of special limits for emergency use or emergency fuels.

(9) Because the various methods of specifying the engine inlet fuel pressure requirements are sometimes related to fuel temperature and altitude, it is often necessary to explore the extremes of the envelope to assure compliance rather than attempting to select one critical condition. Additionally, the rapid increase in fuel viscosity at colder temperatures, which tends to significantly increase system pressure drop, can more than offset a slight drop in required fuel flow such that the critical fuel inlet conditions may not be experienced at maximum engine fuel flow.
Figure AC 27.955–1 illustrates the point.

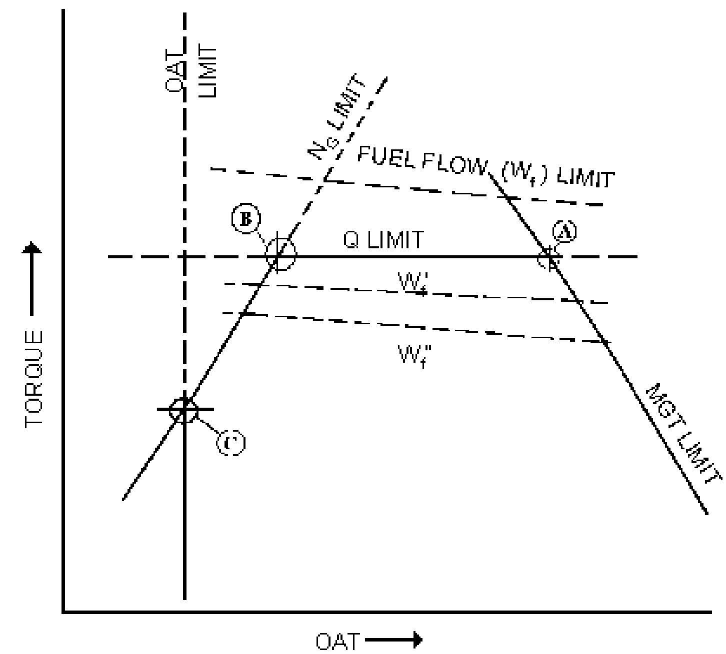


FIGURE AC 27.955-1 FUEL FLOW

NOTES:

(1) Point A on figure AC 27.955-1 is the highest fuel flow within aircraft limitations, but the system pressure drop is not expected to be maximum because of the low kinematic fuel viscosity.

(2) Point B is the maximum flow at cold temperatures but as the fuel temperature is further reduced, the fuel viscosity increases very rapidly.

(3) Point C represents the maximum viscosity of the fuel, but the fuel flow is somewhat reduced from point B. The maximum system pressure drops and, therefore, minimum fuel inlet pressure may occur between points B and C depending on the specific relationship of fuel viscosity to required fuel flow.

(10) A conservative demonstration would consider the maximum allowable fuel viscosity in combination with the maximum fuel flow. Otherwise, several test points may be required.

(11) For those systems which specify a minimum V/L ratio, the methods provided in Aerospace Recommended Practice (ARP) 492 published by the Society of Automotive Engineers are acceptable in evaluating test results.

(12) Since the lower quantity of fuel in the tank will reduce the hydrostatic head and thus the fuel inlet pressure, § 27.955(a)(2) specifies that the quantity of fuel in the tank should be minimum.

(13) Section 27.955(a)(4) specifies that each main and emergency pump be evaluated. If it can be determined which pump and flow path is critical, only that configuration would be tested. Similarly, for suction fuel systems, the critical flow paths and flow requirements should be evaluated. If pumps are required to supply the necessary fuel, § 27.1305(c) would require a fuel pressure indicator and § 27.1549 would require a red radial at the minimum safe operating fuel pressure for any fuel or fuel usage condition. This pressure limit should be used to determine compliance with § 27.955(a)(1) for all operations.

(14) Section 27.955(b) requires the fuel system to feed promptly when fuel becomes low in one tank and another tank is selected. This requirement is important because momentary fuel flow interruption must be expected to result in complete power failure and, for single engine rotorcraft, an emergency landing.

AC 27.955A. <u>§ 27.955 (Amendment 27-23) FUEL FLOW</u>.

a. <u>Explanation</u>. Amendment 27-23 adds new requirements for test conditions to ensure that adequate fuel flow is available to the engine in critical combinations of adverse conditions that may be expected during operation of the rotorcraft. The amendment also requires a correlation between fuel filter blockage and the fuel filter warning device required by § 27.1305(q). Design and performance standards for auxiliary fuel tank and transfer tank fuel systems are provided. These changes were made to ensure that all parameters associated with fuel supply to the engine are adequately addressed.

b. <u>Procedures</u>. All of the policy material pertaining to this section remains in effect with the following additions:

(1) Section 27.955 is intended to ensure adequate fuel flow to the engine(s) during all operating conditions of the rotorcraft. This includes the fuel flows necessary to operate the engine(s) under the test conditions required by § 27.927. Testing (including bench or rig tests) has been the accepted method of showing compliance with this section although analytical techniques may be used to adjust system test results to various fuel flow conditions or to account for minor modifications to a system.

Analytical methods that are used to adjust the test results should be verified with limited testing. It should be shown during compliance testing that the fuel pressure, at the engine to airframe interface, will be within the limits specified by the engine manufacturer. The fuel pressure at this point should be maintained within limits specified by the engine manufacturer during all critical maneuvers and accelerations. All of the following conditions should be met during compliance testing unless it can be shown that combinations of the conditions are not possible.

(i) The fuel quantity in the tank(s) in use during the test may not exceed the unusable fuel quantity established under § 27.959, plus the minimum quantity required to conduct the test.

(ii) During the compliance test, the rotorcraft should be maneuvered to create the most critical fuel pressure head between the fuel tank outlet and the engine to airframe interface (engine fuel inlet).

(iii) For boost pump fed systems, it should be determined which pump (primary or secondary) would create the most critical restriction if it failed. The critical pump should then be installed to create the critical restriction, either by actual or simulated failure.

(iv) Various combinations of engine power demand, electrical power available, and motive flow requirements for ejector pumps, will have an effect upon the fuel flow and pressure available at the engine to airframe interface. Adequate fuel pressure should be available to the engine with the most critical combination of these parameters.

(v) Critical values of fuel properties that may adversely affect fuel flow and/or fuel pressure should be applied. This includes alternate types of fuel if certification with alternate fuels is requested. At the minimum, the fuel that will create the highest vapor to liquid ratio should be used during hot fuel tests (§ 27.961). The most viscous fuel should be used during cold fuel tests.

(vi) The fuel filter, required by § 27.997, should be partially blocked to simulate the maximum contamination allowable. The blockage should be sufficient to activate the impending bypass indicator that is required by § 27.1305(q).

(2) Unique Conditions. The phrase, "…Provide the engine with at least 100 percent of the fuel required under all operating and maneuvering conditions…" (§ 27.955(a)), includes unique flight conditions within the operational envelope of the rotorcraft. Critical conditions of fuel flow to the engine(s) may exist under the following conditions (and others identified by the applicant); therefore, they should be evaluated and tested if applicable:

(i) In a single engine rotorcraft, a rapid acceleration to maximum power (torque) that will be requested for certification may be a critical condition. In this case the fuel flow required during the transient may exceed the fuel flow required for steady state at the maximum power condition.

(ii) In multiengine rotorcraft, a rapid acceleration to the maximum OEI power rating that will be requested may be a critical condition. The fuel flow during the transient may be higher than that required at the steady state OEI condition.

(3) If auxiliary fuel pumps (boost pumps) are used to supply fuel to the engines, and ejector pumps are used for cross-feed or other inter-tank fuel distribution systems, a test should be run that will place the maximum fuel demand on the auxiliary pump(s).

(4) In some multiengine rotorcraft, a single pump may be required to provide fuel flow to all engines in the event of an auxiliary pump failure. If this is the case, a test should be conducted with a simulated (or actual) failed auxiliary pump. If the functional auxiliary pump is designed to provide motive flow for cross-feed systems, the most critical condition of fuel flow demand should be tested.

(5) Transient and steady state maneuver loads (g-loading) may affect the fuel pressure at the engine to airframe interface. This effect should be considered and then tested, if appropriate.

(6) The methods of specifying the engine inlet fuel pressure requirements are sometimes related to fuel temperature and altitude. Therefore, it is necessary to explore the extremes of the envelope to assure compliance rather than attempting to select one critical condition. For instance, the increase in fuel viscosity at cold temperatures may increase system pressure drop and offset a slight drop in required fuel flow. In this case, critical fuel inlet conditions may not be experienced at maximum engine fuel flow.

(7) A conservative demonstration would consider the maximum allowable fuel viscosity in combination with the maximum fuel flow. Otherwise, several test points may be required.

(8) <u>Fuel Transfer Systems</u>. Section 27.955(b) specifies that if normal operation of the rotorcraft fuel transfer system continually delivers fuel to an engine feed tank, and maintains a specific fuel level in the feed tank, then the specified fuel level in the feed tank should be maintained automatically during all flight or surface operating conditions expected with the rotorcraft.

AC 27.959. <u>§ 27.959 UNUSABLE FUEL SUPPLY</u>.

a. <u>Explanation</u>. This rule requires the applicant to establish a value for unusable fuel for each tank. This value for unusable fuel may be selected by the applicant to facilitate compliance with § 27.1337(b)(1) provided the amount is equal to or greater than the actual unusable fuel. The actual unusable fuel is the amount of fuel in the tank when, in the critical flight attitude, evidence of system or engine malfunction occurs or, in the case of transfer tanks, when flow to the receiving tank is interrupted.

b. Procedures.

(1) The unusable fuel for each tank can be determined by flight tests which involve flight in the critical stable attitude and during maneuvers until indication of a malfunction. Maneuvers should be conducted to be critical or conservative with respect to unusable fuel. For boosted systems, the "first evidence of malfunction" may be a pressure fluctuation to below the fuel pressure minimum redline, engine power fluctuation, or boost pump failure warning indication. For suction lift systems, the indication may be engine power interruption. Since an accurate measurement of the remaining fuel in the tank should be obtained, a method to close off flow from that tank would be needed. For transfer tanks, or tanks which are limited to use only during cruise flight, the flight regimes usually can be limited to level flight at the CG condition which, by inspection, would create the maximum unusable fuel. For tanks for general use, the flight regimes should also include takeoff and landing using pitch attitudes to be expected, as well as hover and level flight conditions. The possible adverse effects of extreme lateral CG should be considered.

(2) Normally, these tests are conducted with all equipment (pumps, ejectors, etc.) operating as prescribed by the design. However, values for unusable fuel with pump failures, if significantly different, should also be determined and listed in the flight manual. The value for unusable fuel to be considered in the empty weight of the aircraft should be that value determined with the pump(s) operating normally; i.e., pump failure need not be considered.

c. While the procedures of paragraph (b)(1) are acceptable, fuel exhaustion during critical flight test conditions must be expected. To minimize this possible flight test hazard, the applicant may, in many cases, utilize analysis and/or ground tests involving normally available flight test data on aircraft attitudes, tank configuration studies, and critical flight condition studies to determine unusable fuel. Any questionable results, however, should be resolved by actual flight test or introduction of conservatism into the finding.

AC 27.961. § 27.961 FUEL SYSTEM HOT WEATHER OPERATION.

a. Explanation.

(1) Section 27.961 specifies that a hot fuel test be conducted on suction lift systems, and on other fuel systems conducive to vapor formation, to ensure that the system is free from vapor lock at a fuel temperature of 110° F under critical operating conditions.

(2) Pressure boosted systems would not ordinarily require hot fuel tests unless-

(i) There are high points in the fuel system which would allow accumulation of vapor; or

(ii) The engine fuel inlet pressure is negative relative to tank pressure because of low boost pump pressure or high fuel system pressure losses (but still within fuel pressure limits).

(3) The requirement to use 110° F fuel is a carryover from the recodification of CAR Part 6, although the use of hotter fuel would tend more toward vapor formation.

(4) The term “vapor lock” means a change in normal engine operation as a result of the formation of fuel vapor-air mixtures in the fuel feed system.

b. Procedures.

(1) The fuel type to be used should be that with the highest true vapor pressure (TVP) at the 110° F condition.

(2) The fuel should be heated as rapidly as possible since the longer fuel is heated the more vaporization occurs resulting in unconservative test results.

(3) If the test is performed at cool ambients, the fuel lines, tanks, etc., may have to be insulated to ensure that the fuel inlet temperature is approximately the same as would be experienced on a hot day.

(4) The fuel level should be the lowest consistent with test safety.

(5) The flight tests to the service ceiling should include maximum power climbs to selected intermediate altitudes where various maneuvers including the following are performed:

(i) Low power descent with rapid transition to takeoff power.

(ii) Turns and cyclic pull-ups with load factors comparable to the flight strain survey.

(iii) For multiengine rotorcraft with 30-minute and/or 2.5-minute OEI power ratings, conduct a rapid single-engine acceleration from low power to engine topping power followed by cruise at the maximum allowable OEI power.

(6) The flight test maneuvers should be repeated at the service ceiling.

(7) Except for transients and descents, the power available used should correspond to a 100° F sea level day lapsed 3.6° F/1,000 foot pressure altitude.

(8) Engine operation throughout the test should be normal; i.e., no surge, stall, flameout, etc., and the engine fuel inlet requirements should not be exceeded.

(9) Alternative tests on appropriate test rigs may be conducted ensuring proper simulation of altitude, ambient temperature, fuel temperature, fuel flow, and load factors.

AC 27.961A. § 27.961 (Amendment 27-23) FUEL SYSTEM HOT WEATHER OPERATION.

a. Explanation. Amendment 27-23 simplifies and restates the fuel system hot weather certification requirements and adds a requirement for the system to be capable of providing adequate fuel during probable transients. These changes clarified the existing wording to assure adequate qualification testing.

b. Procedures. This paragraph specifies that all suction lift systems and any other fuel system that may be conducive to vapor formation show satisfactory engine fuel inlet conditions (within criteria established by the engine manufacturer) when using the fuel with the highest true vapor pressure (TVP) at 110° F fuel temperature. Engine operating conditions should include those defined by §§ 27.927(b)(1) and 27.927(b)(2). Compliance can be shown by analysis, testing, or a combination of both.

AC 27.963. § 27.963 FUEL TANKS: GENERAL.

a. Explanation.

(1) Paragraph (a) sets forth general requirements for fuel tank structural aspects.

(2) Paragraph (b) requires design features to react forces to be expected from fuel surging due to accelerations of the rotorcraft.

(3) Paragraph (c) requires design features to ensure heat transfer from an engine compartment fire will not jeopardize the fuel tank integrity.

(4) Paragraph (d) requires design features to minimize the hazards of a leaking fuel tank and also requires design features to ensure that unwanted transfer of fuel from one tank to another does not occur due to differences of pressure in the tanks.

b. Procedures.

(1) For paragraph (a), the tests of § 27.965 are normally adequate if performed in conjunction with the reliability test of § 21.35 or other service simulation tests.

(2) For paragraph (b), internal or external stiffening may be required for surge resistance. If the analysis provided to show the adequacy of the surge resistance is

questionable, the slosh and vibration tests of § 27.965 may be accepted as substantiation of this requirement.

(3) The fuel tank clearance required by paragraph (c) may be determined by inspection of the design.

(4) The ventilation and interconnect requirements of paragraph (d) may usually be determined by flight tests which explore maximum rates of climb and descent with sensitive pressure measuring equipment installed inside tanks and in the ventilation airspaces provided to comply with this rule.

AC 27.963A. § 27.963 (Amendment 27-23) FUEL TANKS: GENERAL.

a. Explanation. Amendment 27-23 added new subsections (e) and (f) that require designs and tests to ensure that no exposed surface inside a fuel tank would, under normal or malfunction conditions, constitute an ignition source. They also set forth standards for the design and qualification of fuel tanks located in personnel compartments. These requirements are needed to ensure freedom from the hazards of fuel tank internal explosions and to ensure that fuel tanks, installed in passenger compartments, present no hazards to the personnel or to the rotorcraft.

b. Procedures. Section 27.963(e) requires the temperature of any exposed surface inside a fuel tank to be at least 50° F lower than the lowest auto-ignition temperature of the fuel or fuel vapors in the tank (reference paragraph AC 27.1185b(3), § 27.1185). For compliance with § 27.963(e), the internal component surface temperatures can be determined by flight or laboratory tests. The most critical flight conditions are established with sensitive temperature and pressure measuring equipment. This equipment is installed inside the tanks and in the ventilation air spaces.

AC 27.963B. § 27.963 (Amendment 27-30) FUEL TANKS: GENERAL.

a. Explanation. Amendment 27-30 adds a new requirement to paragraph (g) that (in addition to the current requirements) requires that the fuel tank bladder or liner be puncture resistant by meeting the TSO C-80, paragraph 16.0, screwdriver test requirements, using a new crash resistance based minimum puncture force of 370 lbs. A new requirement is also added to paragraph (f). Paragraph (f) now additionally requires that each fuel tank installed in a personnel compartment be crash resistant by meeting the applicable criteria of the new Crash Resistant Fuel System requirements of § 29.952 (Re: paragraph AC 27.952).

b. Procedures.

(1) Paragraph (g). The procedures for old paragraph (g) still apply under new (g). In addition, to comply with the added puncture resistance requirement under new (g), the requirements of § 27.952(h) must be met. Paragraph AC 27.952 gives the

detailed compliance procedures for § 27.952(h). The compliance procedures for § 27.952(h) also provide compliance for puncture resistance under § 27.963(g).

(2) Paragraph (f).The procedures for old paragraph (f) still apply under new (f). Compliance with the added crash resistance requirement of new (f) can be shown by conducting a thorough design review of each fuel tank compartment to ensure that all the regulatory criteria are met. (All fuel drains and vents should also be reviewed to ensure that they meet applicable § 27.952 requirements.) A basic static loads analysis followed by a stress analysis is typically used to determine that the enclosure protects the fuel tank and provides the crash resistance level necessary for occupant survival in an otherwise survivable impact. The applicable emergency load factors are typically used to design the enclosure. (Section 27.952 contains the corresponding load factors for fuel cells and their attachments.) The emergency load factors are typically adequate for all loading conditions encountered by the enclosure in service. The typical design approach is to design the enclosure to crush at a rate approximately the same as the crush rate of the fuel tank and to ensure that all puncture hazards (such as sharp projections either enhanced or created by impact that would penetrate the fuel tank) are minimized in design. (See paragraph AC 27.952 guidance material for details.)

27.965. § 27.965 (Amendment 27-12) FUEL TANK TESTS.

a. Explanation. This regulation defines the tests that must be accomplished to show compliance for rotorcraft fuel tanks.

(1) Four basic types of fuel tanks are: (1) a metal tank installed in the aircraft or at the wing tip; (2) an integral tank; (3) a nonmetallic self-supporting tank (fiberglass); and (4) nonmetallic flexible bladder-type tanks.

(2) There are two basic tests required by the regulations. One test procedure substantiates the design by tests and analysis by applying applicable pressure to the tank. The other procedure substantiates the design by vibration and slosh tests of the tanks.

b. Procedures.

(1) Pressure Test. The 3.5 or 2.0 PSI pressure test listed in the regulations should be conducted unless the pressure with a full tank for maximum limit acceleration or emergency acceleration is greater. Section 27.337 gives the value for the limit acceleration.

(2) Vibration and Slosh Tests.

(i) There is not an absolute value of what constitutes “large” unsupported or unstiffened flat areas. However, it has generally been considered that any fuel tank with less than 10 gallons capacity, constructed with simple, wide, flat geometric shape and using metal (in metal tanks) of 0.05-inch thickness or greater would not require tests in accordance with § 27.965(d). Using this basis, a 14- by 14-inch properly constructed tank would not require vibration and slosh tests.

(ii) If the tank construction is of a metal or integral design which can be shown to be similar to previously approved tanks with acceptable service history, the vibration and slosh tests may not be required. Similarity would entail comparing the construction technique; i.e., similar panel size, similar sealing methods, skin and angle thickness, loads being similar, etc.

(iii) For fuel tanks located in the sponson or stub wing, the entire sponson or wing should be rocked and vibrated unless it can be determined that a certain portion of the tanks is critical. In this case a fixture should be developed such that the portion of the tank being tested is rocked about a pivot point which would produce the same amplitudes of motion for the portion of the tank being tested, as if the whole sponson or wing was being tested. Structure loads in conjunction with these tests have not been required.

(iv) The amplitude of vibration specified in the regulation is double amplitude (peak to peak). Vibration amplitudes less than one thirty-second of an inch must be justified by instrumented tests of the tank installed in the aircraft.

(v) The vibration and slosh procedures listed in Military Specification, MIL-T-6396, have been accepted to show compliance with § 27.965(d).

(3) After all tests have been conducted, the tanks should be leak checked using test fluid conforming to Federal Specification TT-S-735 type III or equivalent.

AC 27.967. <u>§ 27.967 FUEL TANK INSTALLATION</u>.

a. <u>§ 27.967(a)</u>:

(1) <u>Explanation</u>. This paragraph was added by Amendment 27-30 to create parallelism of both regulatory structure and level of safety between Parts 27 and 29 by the Crash Resistant Fuel System final rule on October 3, 1994. This paragraph sets forth a series of detail requirements for fuel tanks intended to ensure that tank leakage or failure is unlikely. These regulatory requirements pertain primarily to proper support of the tank and protection against chafing.

(2) <u>Procedures</u>. For conventional metal tanks, the support devices, commonly called “cradles,” should be designed with wide flanges or cap strips at the contact area with the tank to distribute the loads in the tank material. To prevent chafing, install nonmetallic padding, treated to eliminate absorption of fuel between the tank and the support structure. Cork strips sealed with shellac and bonded to the support structure have been found suitable. Fuel cell sealant material should be applied over rivet heads and in corners. Bladder cells must be designed to fit accurately in the cell cavity in

order to avoid fluid loads in the bladder itself. The interior of the cavity should be smooth to avoid damage to the bladder cells.

b. § 27.967(b):

(1) Explanation. This paragraph requires the design to provide ventilation and drainage of spaces adjacent to fuel tanks to avoid accumulation of fuel or fumes to be expected from minor leakage of fuel tanks. This is needed to minimize the possibility of fire or explosion in these spaces. An exception to this requirement is allowed for bladder cells installed in a closed compartment. For this configuration, ventilation may be limited to that provided by compartment drains if the ventilation is adequate to maintain proper pressure relationship between the bladder cell and cell compartment air spaces.

(2) Procedures. With the assumption that fuel tank leakage will occur, require the tank compartments to be provided with drains at any low point. These drains should conduct fuel clear of the rotorcraft and should be three-eighths of an inch or larger in diameter to minimize clogging. As with any drain intended to function in flight, verification that reverse flow will not occur due to pressure differentials at each end of the drain is appropriate. Ventilation for these tanks should involve openings in the compartment such that in-flight slipstream and/or rotor downwash will rapidly and continuously purge the tank compartment of fuel fumes. Openings should not be located so the fumes or fuel can reenter the rotorcraft. For flexible tank liner configurations (bladder cells), no specific ventilation is required if the cell is located in a compartment which is closed, except for drain holes. Note that a cell leak may be expected to produce fumes in the compartment airspace which are flammable; thus, items installed in bladder tank cavities shall not create a hazard during either normal or malfunction conditions. The vent system for the interior of the cell must be adequate to ensure that the bladder cell interior pressure is always positive or at least neutral with respect to any other airspace in the cell compartment to prevent collapse of the bladder cell. Drainage of the cell compartment should meet the criteria discussed above.

(3) A light mesh or string network hung between the bladder cell and its compartment walls is recommended to provide seepage channels to facilitate fuel leakage to the low-point compartment drains.

c. § 27.967(c):

(1) Explanation. This paragraph requires a measure of protection for fuel tanks from adverse effects of a fire in a fire zone.

(2) Procedures. Verify that a firewall meeting the requirements of § 27.1185 effectively separates any fuel tank from any engine. To minimize hazards of heat transfer to a fuel tank through a fire wall during an engine compartment fire, verify that at least one-half inch of clear airspace exists between the tank and the firewall.

d. § 27.967(d):

(1) Explanation. This paragraph is intended to prevent hazards to integral fuel tanks to be expected by impingement of flames or products of combustion from an engine compartment fire.

(2) Procedures. Review the design for relative positions of engine compartments and integral fuel tanks to estimate the flowpath of fire or heat from an engine compartment fire. Consider autorotation for single-engine rotorcraft and, for multiengine rotorcraft, low power descent as power-on flight in this evaluation. If questionable compliance exists, clear indication of the flow impingement patterns may be identified by ejecting dye from engine compartment openings during flight.

AC 27.969. § 27.969 FUEL TANK EXPANSION SPACE.

a. Explanation.

(1) Space must be provided in each fuel tank system to allow for expansion of the fuel as a result of a fuel temperature increase. The space provided for this purpose must have a minimum volume equal to 2 percent of the tank capacity.

(2) The fuel tank filling provisions must be designed to prevent inadvertent filling of the fuel tank expansion space when fueling the rotorcraft in the normal ground attitude on level ground.

b. Procedures.

(1) Fuel tanks with interconnected vents need not have provisions for fuel expansion in each tank if equivalent expansion provisions are available in another area.

(2) The fuel filler ports should be located below the designated fuel expansion space height to ensure that the fuel expansion space cannot be inadvertently filled with fuel.

(3) Each fuel tank expansion space must comply with the venting requirements of § 27.975.

(4) For multiengine rotorcraft using a single expansion tank to satisfy the requirements of this regulation, the effect of blockage or failure of any vent from this common tank must be considered with respect to compliance with the applicable engine isolation requirements.

AC 27.969A. § 27.969 (Amendment 27-23) FUEL TANK EXPANSION SPACE.

a. Explanation. Amendment 27-23 allows some interconnected fuel tanks to have a common expansion space in lieu of individual expansion spaces. This change relieves complex design requirements where simpler designs have proven to be satisfactory.

b. Procedures. There is no change to the suggested methods of compliance.

AC 27.971. § 27.971 FUEL TANK SUMP.

a. Explanation.

(1) Each fuel tank must be provided with a drainable sump which is located at the lowest point in the tank with the rotorcraft in a normal ground attitude.

(2) The main fuel supply to any engine may not be drawn from the bottom of any fuel sump.

(3) Each fuel sump drain must comply with the requirements of § 27.999.

b. Procedures.

(1) Each fuel sump should have an effective capacity which is not less than 0.25 percent of the tank capacity or 1/16 gallon, whichever is greater, with the rotorcraft in any ground attitude to be expected in service. This sump capacity will provide a level of safety equivalent with other normal category aircraft (reference § 23.971).

(2) Demonstration of compliance with the minimum sump capacity requirements may be shown by analysis, test, or a combination of both depending on the complexity of the fuel system design.

(3) If minimum sump capacity is to be demonstrated by test, the following general test procedures will produce acceptable results:

(i) Determine the most critical ground attitude to be expected in service from such considerations as uneven terrain, slope landing limits, etc. The critical attitude for each tank will be that for which the maximum amount of fuel can be withdrawn from the tank using the rotorcraft's fuel supply system.

(ii) Using a rotorcraft with a fuel system which conforms to the final design specification, position the rotorcraft to the critical attitude for the tank to be tested using leveling jacks, actual terrain of a predetermined slope, or other similar means.

(iii) Using the rotorcraft's fuel supply system, pump fuel from the tank being tested until the supply system will no longer withdraw fuel. This can be done without the rotorcraft engine actually running unless an engine driven pump is an essential component of the fuel supply system. Caution should be exercised if an engine is to be run to fuel exhaustion since engine surge at the pump cavitation point can result in damaging torsional loads in the transmission drive system.

(iv) When no more fuel can be removed from the tank with the rotorcraft fuel supply system, return the rotorcraft to a normal ground attitude. Completely drain the sump of the tank or tanks being tested into a container and measure the volume drained from each sump. The volume measured must satisfy the minimum capacity requirements of paragraph AC 27.971b(1).

AC 27.971A. <u>§ 27.971 (Amendment 27-23) FUEL TANK SUMP</u>.

a. <u>Explanation</u>. Amendment 27-23 prescribed minimum values for fuel tank sump capacity, authorized the use of a sediment bowl in lieu of a sump, and required these sumps or sediment bowls to be effective in any ground attitude which can reasonably be expected in service.

b. <u>Procedures</u>. The policy material pertaining to this section remains in effect. Additionally, if the rotorcraft is equipped with a sediment bowl or chamber, the capacity should be at least one ounce for every 20 gallons of fuel tank capacity. The sediment bowl or chamber should be located so that water will drain from all parts of the tank to the sediment bowl or chamber when the rotorcraft is in any allowable normal ground attitude. Compliance with the minimum sump capacity or the sediment bowl or chamber requirements may be shown by analysis, test, or a combination of both, depending upon the complexity of the fuel system.

AC 27.973. <u>§ 27.973 FUEL TANK FILLER CONNECTION</u>.

a. <u>Explanation</u>. Fuel tank filler connections must be designed so that no fuel can enter into any part of the rotorcraft other than the fuel tank during fueling operations. Spilled fuel must be considered as well as fuel entered into the fuel filler port.

b. <u>Procedures</u>.

(1) Each fuel filler opening must be identified with the markings and placards required by § 27.1557.

(2) Each filler cap should provide a fuel-tight seal for the main filler opening unless the fuel tank is vented through a small opening in the filler cap.

(3) Each fuel filling point should have a provision for electrically bonding the rotorcraft to ground fueling equipment.

(4) Compliance with the requirements of this paragraph can normally be demonstrated by analysis and physical inspection of the fuel filler design. Testing is not normally required.

AC 27.973A. <u>§ 27.973 (Amendment 27-30) FUEL TANK FILLER CONNECTION</u>.

a. <u>Explanation</u>. The original, single unlettered paragraph of old § 27.973 is redesignated as paragraph (a) by Amendment 27-30. The new (a) has three subparagraphs. These changes have been made to both make § 27.973 parallel to § 29.973 and to incorporate the new crash resistant fuel system requirements of § 27.952 (re: paragraph AC 27.952).

(1) New paragraph (a) is revised to require that all fuel tank filler connections be made fuel tight under both normal operations and during a survivable impact in accordance with the requirements of § 27.952(f) and its associated advisory material.

(2) New paragraph (a)(1) is added to require that each filler be marked as prescribed in § 27.1557(c)(1).

(3) New paragraph (a)(2) is added to require that each recessed filler connection that can retain an appreciable amount of fuel have a drain that discharges clear of the rotorcraft.

(4) New paragraph (a)(4) is added to require that each filler cap provide a fuel tight seal under the fluid pressures expected in service and in a survivable impact.

(5) New paragraph (b) is added to require that each filler cap or cap cover warn when the cap is not fully locked or seated to a fuel tight condition on the filler connection.

b. <u>Procedures</u>.

(1) The compliance procedures for general paragraph (a) are those of § 27.952(f) and those described herein for the three subparagraphs to (a).

(2) The compliance procedures for (a)(1) and (a)(2) can normally be demonstrated by analysis and physical inspection of the fuel filler design. Testing is not normally required.

(3) The compliance procedures for (a)(3) are as follows: The fuel tank filler connection must be shown to be leak free under the worst case fuel pressures (due to combination of static pressure and sloshing induced head) from both normal operations and from a survivable impact. The worst case loads from these two conditions must be determined. In most cases the load resulting from a survivable impact will prevail. For the survivable impact, normally the worst case combined pressure loading occurs at the time of impact at the fuselage that places the filler tube neck (at the vicinity of the filler cap connection) in a vertical or near vertical attitude. Once the critical load case is determined by analysis, test, or a combination; the fuel tank filler connection (or an approved mockup) can be tested for sealing capability by applying a fluid such as water at the critical pressure at the critical attitude of the tube (with the cap inverted) for a

period of at least 5 minutes. If no significant leakage occurs, then compliance has been shown. Significant leakage is defined as leakage in excess of 10 drops per minute at any time during or after the 5-minute test.

(4) Compliance procedures for paragraph (b) are as follows: Visual means, such as placards and alignment marks, and mechanical means, such as detents and locking slots, must both be provided. This is necessary to give both a clear visual and mechanical indication that a filler cap or a filler cap cover is properly installed and fuel tight after each removal and replacement. Visual indications such as alignment marks, that show proper installation should be easily read from a distance of at least 5 feet by anyone making a routine inspection or check.

AC 27.975. § 27.975 FUEL TANK VENTS.

a. Explanation.

(1) Each fuel tank for which an expansion space is required per § 27.969 must be vented from the top part of the expansion space.

(2) Fuel tank vents must be designed to minimize the probability of the vent being restricted or completely clogged by dirt or ice.

(3) Vents of fuel tanks having interconnected outlets must be interconnected as required per § 27.963.

b. Procedures.

(1) There should be no point in any vent line where moisture can accumulate with the rotorcraft in the ground attitude or level flight attitude unless drainage is provided.

(2) Each vent should be constructed to prevent siphoning of fuel during any normal operation.

(3) No vent line or drainage provision should be terminated at a point where the discharge of fuel from the outlet would constitute a fire hazard or from which fumes could enter any personnel compartment.

(4) The vent system capacity and installed configuration should maintain acceptable differences of pressure between the interior and exterior of tank. Analysis and/or flight testing may be required to demonstrate this capability depending on the fuel system design. If flight testing is required, the following flight test procedure is one method of verifying proper vent system operation.

(i) Using a rotorcraft with a fuel tank and vent system which conforms to production design specifications, install differential pressure instrumentation which will measure the difference between the gas pressure inside each fuel tank expansion space and the air pressure in the cavity or area surrounding the outside of the fuel tank.

(ii) Conduct ground and flight tests recording the differential pressures between the inside and the outside of the fuel tanks. The following conditions should be evaluated:

(A) Refueling and defueling (if applicable).

(B) Level flight to V_{NE}.

(C) Maximum rate of ascent and descent.

(iii) Compare the measured differential pressure values with the maximum allowable for the fuel tank design being evaluated. For flexible bladder type fuel cells, the pressure inside the tank should not be significantly less than the surrounding pressure to avoid the possibility of collapsing the bladder.

AC 27.975A. § 27.975 (Amendment 27-23) FUEL TANK VENTS.

a. Explanation. Amendment 27-23 added a new paragraph § 27.975(b) that requires fuel tank vent systems be designed to minimize fuel spillage and subsequent fire hazards in the event of rollover of the rotorcraft during landing or ground operation.

b. Procedures. The policy material pertaining to this section remains in effect. Additionally, fuel tank vent system design should minimize spillage of fuel in the vicinity of a potential ignition source in the event of rollover during landing or ground operation.

AC 27.975B. § 27.975 (Amendment 27-30) FUEL TANK VENTS.

a. Explanation. In addition to the current requirements, Amendment 27-30 revises paragraph (b) to add the requirement that the venting system be designed to minimize fuel spillage through the vents to an ignition source in the event of a fully or partially inverted rotorcraft fuselage attitude following a survivable impact. (A survivable impact is defined in paragraph AC 27.952.) Since rotor action on impact and other impact dynamics have been found in numerous cases to cause rollovers or other unusual postcrash attitudes, compliance with this paragraph would significantly mitigate the postcrash fire hazard by minimizing fuel spills through vents to ignition sources when the postcrash attitude of the rotorcraft would allow gravity and/or post impact sloshing induced fuel spills through a normally open fuel vent.

b. Procedures

(1) In addition to the compliance procedures for the previous amendment; installation of design features, such as gravity activated shuttle valves in the vent lines (that are normally open but close under certain predictable, postcrash scenarios that are

generated by involvement in a survivable impact that results in either an inverted or partially inverted fuselage attitude) must be accomplished.

(2) Once selected, the design feature chosen for compliance should be shown to function effectively without significant leakage by either full scale and/or bench tests that apply the total pressure forces that correspond to a 100 percent full, 50 percent full, and 5 percent full fuel load applied to the device in a worst case survivable impact. (If a critical fuel level can be clearly identified, then only that fuel level and the corresponding critical total pressure load need be utilized for certification approval.) The total pressure forces should be determined and applied in a manner that simulates the magnitude and rate of load onset (due to a combination of gravity and sloshing) that would occur in otherwise survivable impacts that would involve rollover attitudes of 45 degrees (or the minimum spillage roll angle), 90 degrees (rotorcraft on its side), and 180 degrees (rotorcraft fully inverted). (In some designs, the 45-degree attitude may not be the correct initial roll angle at which fuel spillage through a given vent would begin to occur due to the placement of the vents on the fuselage. For these cases, the minimum angle should be determined by analysis.)

(3) Once all test conditions are defined, these tests should be conducted with all structural deformation present in the test set up that is necessary to simulate the actual structural deformation either in or applied to the vent line or system in a worst case survivable impact. The structural deformation to be applied can be determined by rational analysis, analysis, test, or a combination. Significant leakage is defined as leakage of 10 drops per minute, or less, after all testing is complete. The criteria of 10 drops per minute, or less, corresponds to the criteria of 5 drops per minute, or less, per breakaway coupling half (i.e., a total of 10 drops per minute, or less, for the entire separated coupling) specified in the advisory material for § 27.952 (re: paragraph AC 27.952).

AC 27.977. <u>§ 27.977 (Amendment 27-11) FUEL TANK OUTLET</u>.

a. <u>Explanation</u>.

(1) This provision prescribes a fuel strainer for the fuel tank outlet (suction lift system) or for the booster pump (boosted systems) for both reciprocating and turbine engine installations.

(2) This requirement is intended to ensure that relatively large, loose objects which may be present in the fuel tank do not interfere with fuel system operation. The provision of § 27.997 should ensure protection from smaller contaminants which may occur in service.

b. <u>Procedures</u>.

(1) Section 27.977(a) specifies an 8- to 16-mesh-per-inch strainer for reciprocating engine installations, and a strainer which will prevent passage of any

object which could restrict fuel flow or damage any fuel system component for turbine installations.

(2) In addition to the requirement of § 27.977(a), the flow area of the strainer should be at least five times the area of the outlet line. Furthermore, the diameter of the strainer must be at least that of the fuel tank outlet line.

(3) Each finger strainer should be accessible for inspection and cleaning.

(4) Compliance with § 27.977 is usually verified by inspection, and testing is not required. The ice protection provisions of § 27.951(c) are applicable to the strainer at the fuel outlet, and testing to show compliance with that provision may be required.

SUBPART E - POWERPLANT

FUEL SYSTEM COMPONENTS

AC 27.991. § 27.991 FUEL PUMPS.

a. Explanation.

(1) Section 27.991(a) provides a definition of the main pump(s) and § 27.991(b) requires an "emergency pump(s)." The main pump(s) that is certified as part of the engine does not fall under § 27.991 requirements. The main pump(s) discussed under § 27.991 should therefore be considered the "main aircraft pump(s)."

(2) The main aircraft pump(s) consists of whatever pump(s) is required to meet engine or fuel system operation throughout the range of ambient temperature, fuel temperature, fuel pressure, altitude, and fuel types intended for the rotorcraft. If the main aircraft pump(s) is required to meet the above criteria, then an emergency pump(s) is required. Airframe supplied pumps intended for use during engine starting only are not considered to be main aircraft pumps and do not require emergency backup pumps.

b. Procedures.

(1) Each pump classified as a main aircraft pump, which is also a positive displacement pump, must have provisions for a fuel bypass. An exception is made for fuel injection pumps used on certain reciprocating engines and for the positive displacement, high pressure, fuel pumps routinely used in turbine engines. The bypass may be accomplished via internal spring check valve and fuel passage or by external plumbing and a check valve. High capacity positive displacement pumps with internal pressure relief and recirculation passages should be checked for overheating if they may be expected to operate continuously at or near 100 percent recirculation.

(2) Section 27.991(b) specifies a requirement for "emergency" pumps to provide the necessary fuel after failure of any (one) main aircraft pump. (Injection pumps and high pressure pumps used on turbine engines are exempt.) To ensure adequate pressure, the "emergency" pump should produce 100 percent of the engine flow requirement. In addition, to allow for pump or fuel system deterioration or possible filter impediments, 125 percent of takeoff flow at minimum pressure should be provided by the "emergency" pump. As stated in this rule, the "emergency" pump must be operated continuously or started automatically to ensure continued normal operation of the engine. For some multiengine rotorcraft, another main aircraft pump may possibly be used as the required "emergency" pump. In this case, the dual role of this pump requires it to have capacity to feed all engines at the critical pressure/flow condition. Availability of fuel flow from this backup pump must be automatic and this function should be verified in the preflight check procedure. The flight or ground crew should be

provided with a means to determine that a main pump failure has occurred so that it can be replaced in a timely manner.

AC 27.991A. § 27.991 (Amendment 27-23) FUEL PUMPS.

a. Explanation. Amendment 27-23 revised § 27.991 to clarify fuel pump redundancy requirements. Redundancy for fuel pump failure includes consideration of both the pump and the pump motivating device.

b. Procedures. All of the policy material pertaining to this section remains in effect with the following clarification: Airframe supplied fuel pumps that are intended for use only during engine starting are not considered as “main” airframe pumps and do not require “emergency” backup pumps.

AC 27.993. § 27.993 (Amendment 27-2) FUEL SYSTEM LINES AND FITTINGS.

a. Explanation. This rule outlines design requirements for fuel system lines.

b. Procedures.

(1) Compliance is usually obtained by employing routing and clamping as described in paragraph 709, Chapter 14, Section 2, of AC 43.13-1A and by monitoring the arrangement throughout the developmental and certification test period. Requirements for approved flexible lines may be resolved by utilizing lines listed as TSO C53a approved for installation in either normal or high temperature areas as appropriate. The service life of TSO C53a approved high pressure fuel hoses is not established by regulation. Service life is determined by the aircraft manufacturers and included in their quality control system which is monitored by the FAA/AUTHORITY.

(2) Verify that adequate clearance exists between lines and elements of the rotorcraft control system at extremes of control travel, including control deflections and, for flexible lines (hoses), possible variations in routing.

(3) Flexible lines inside fuel or oil tanks require special evaluation to ensure that the external surfaces of these lines are compatible with the fluids involved and that fluid sloshing will not cause line failure. Lines inside tanks should be routed to avoid impingement by fuel or oil filler nozzles.

(4) Fuel system lines and fittings located in any area subject to engine fire conditions must comply with the requirements of § 27.1183.

(5) Compliance with § 27.999 requires that fuel system lines contain no low points from sagging or looped routing unless drains are provided which will completely drain the system with the rotorcraft in its normal attitude on level ground.

(6) Good design practice suggests that all flammable fluid lines should be routed to minimize the possibility of rupture in the event of a crash or from engine rotor disc failure.

AC 27.995. <u>§ 27.995 FUEL VALVES</u>.

a. <u>Explanation</u>. Valves must be provided in the fuel supply system to each primary and auxiliary powerplant which will permit positive fuel flow feeding and shutoff from each fuel supply source. Although the engine throttle control system will provide one positive fuel shutoff means at the engine fuel control, additional fuel shutoff valves will normally be required in each fuel supply system to satisfy the requirements of paragraph (d) of this rule and § 27.1189(c).

b. <u>Procedures</u>.

(1) The fuel valve control must be located within easy reach of the appropriate crewmember and must satisfy the requirements of §§ 27.1141(c) and 27.1189(b).

(2) If independent fuel supply sources are provided, the fuel valve or valves must allow independent feeding and shutoff of fuel from each supply source.

(3) Multiengine rotorcraft fuel systems must have fuel valves which comply with the requirements of § 27.953(b)(1).

(4) No fuel valve may be located on the engine side of any firewall. Each valve should be supported so that loads resulting from its operation or from accelerated flight conditions are not transmitted to the lines connected to the valve.

(5) If check valves are included in the fuel supply system, each check valve should be constructed, or otherwise incorporate provisions, to preclude incorrect installation of the valve.

AC 27.997. <u>§ 27.997 (Amendment 27-20) FUEL STRAINER OR FILTER</u>.

a. <u>Explanation</u>. This rule provides for a main in-line fuel filter designed to collect all fuel impurities which could adversely affect fuel system and engine components downstream of the filter. The rule also requires a sediment bowl and drain (or that the bowl be removable for drain purposes) to facilitate separation of contaminates, both solid and liquid, from the fuel. This section is not intended to require installation of the filter between the fuel tank outlet and the first fuel system component which is susceptible to restricted fuel flow because of contaminates (such as a fuel heater or ice trap equipment).

b. <u>Procedures</u>.

(1) The filter should be mounted in a horizontal segment of the fuel line to facilitate proper action of the sediment bowl. If the filter is located above the fuel tank, it becomes necessary to activate a fuel boost pump to achieve positive drainage of the filter bowl. Without pump pressure, air may enter the fuel system during the filter draining operation and, for turbine engines, result in transient power surges or engine failure during subsequent engine operation. A flight manual note to require pump(s) to be "on" during filter draining would be appropriate.

(2) Section 27.997(d) sets forth a requirement for filter capacity. The capacity requirement may be substantiated by showing that the filter, when partially blocked by fuel contaminates (to a degree corresponding to the indicator marking or setting required by § 27.1305(a)), does not impair the ability of the fuel system to deliver fuel at pressure and flow values established as minimum limitations for the engine. The filter mesh must be sized to prevent passage of particulate matter which cannot be tolerated by the engine. Part 33 requires that the degree and type of filtration be established for the engine. This information, available in the FAA/AUTHORITY-approved Engine Installation Manual, should be the basis for selection of the airframe filter mesh. Although a test may be devised and conducted, data from the filter manufacturer usually are acceptable to verify compliance. Note that when the filter capacity is reached, continued flow of contaminated fuel may result in engine failure. A flight manual note regarding precautionary procedures is appropriate.

(3) Part 33 (through Amendment 33-6) has an identical requirement for a fuel filter for engine fuel systems; however, it is not intended that two filters should be required.

AC 27.997A. <u>§ 27.997 (Amendment 27-23) FUEL STRAINER OR FILTER</u>.

a. <u>Explanation</u>. Amendment 27-26 requires that a fuel strainer or filter should be installed between the fuel tank outlet and the first fuel system component that is susceptible to fuel contamination. Components that will be protected from contamination include but are not limited to fuel metering devices which control flow rate, fuel heaters, and positive displacement pumps. The amendment also requires a sediment bowl and drain (unless the bowl is readily removable for drain purposes) to facilitate separation of solid and liquid contaminants from the fuel.

b. <u>Procedures</u>.

(1) The fuel strainer or filter should be accessible for draining and cleaning. It should incorporate a screen or other element that is easily removable. It should be mounted so that its weight is not supported by the inlet or outlet connections of the strainer itself, unless it can be shown that adequate strength margins exist in the lines and connections.

(2) The fuel strainer or filter should have a sediment trap and drain (unless the trap is readily removable for drain purposes). The volume capacity of the sediment trap is specified in § 27.971(a) (0.10 percent of the tank capacity or 1/16 of a gallon).

(3) The fuel strainer or filter mesh should provide the filtration stipulated in the FAA/AUTHORITY-approved engine installation manual that is prepared for the type certificated engine (FAR Part 33).

(4) The fuel strainer or filter should have the capability to remove any contaminant that would jeopardize the flow of fuel that is necessary to meet the requirements of § 27.955. In addition, the strainer or filter should have a bypass system with an impending bypass indicator (Refer to § 27.1305(a)(17)). When the strainer or filter is partially blocked with contaminants, to the degree that the fuel flow requirements of § 27.955 can no longer be achieved, the impending bypass indicator should be activated. At this point, the strainer or filter should not yet be bypassing unfiltered fuel. Although a test may be devised and conducted, data from the filter manufacturer usually are acceptable to verify compliance. Note that when the filter capacity is reached, continued flow of contaminated fuel may result in engine failure. A flight manual note regarding precautionary procedures is appropriate.

(5) Section 33.67(b) has an identical requirement for a fuel filter for engine fuel systems; however, it is not intended that two filters should be required.

AC 27.999. <u>§ 27.999 (Amendment 27-11) FUEL SYSTEM DRAINS</u>.

a. <u>Explanation</u>. This regulation provides for fuel system drains and defines the requirements which the system must meet.

b. <u>Procedures</u>.

(1) The location and function of the fuel system drains are an integral part of any fuel system. There may be several drains required dependent upon the fuel system design. Each fuel tank sump and certain types of fuel strainers or filters require a means to drain (reference §§ 27.971 and 27.997).

(2) Selection of the location and orientation of the drain discharge in the design phase is important to assure that there is no impingement on any part of the rotorcraft. To show compliance with the requirement may require tests dependent upon whether the applicant has a previously approved design which is similar or if the system is a new design for which no previous experience is available.

(3) The location of the drain valve should be selected so that the requirements for accessibility, ease of operation, and protection are met.

(4) Spring-loaded fuel drain valves conforming to MIL-V-25023B, TSO-C76, or equivalent, may be approved as "positive locking" valves for those installations where

the person operating the valve can visually confirm that the valve is closed, provided the applicant has shown that the valve will not open inadvertently under any foreseeable operating condition.

AC 27.999A. § 27.999 (Amendment 27-23) FUEL SYSTEM DRAINS.

a. Explanation. Amendment 27-23 adds the requirement that fuel system drains be effective with the rotorcraft in any allowable ground attitude including uneven terrain. In addition, the change amended § 27.999(b)(2) to require fuel drains have a means to ensure positive closure, as contrasted to positive locking, when in the “off” position. This will accommodate designs featuring spring-loaded drain closures that have been found to be satisfactory.

b. Procedures. All of the policy material pertaining to this section remains in effect. Additionally, selection of the location and orientation of the fuel drain discharge in the design phase is important to assure that there is no impingement upon any part of the rotorcraft. The location and orientation should also ensure effective fuel drainage when the rotorcraft is parked on uneven terrain. To show compliance with the requirement, tests may be required, dependent upon whether the applicant has a previously approved design that is similar, or the system is a new design for which no previous experience is available.

SUBPART E - POWERPLANT

OIL SYSTEM.

AC 27.1011. § 27.1011 (Amendment 27-23) OIL SYSTEM--GENERAL.

a. Explanation.

(1) This regulation defines the general oil system requirements for the engine.

(2) Each engine oil system should be independent of the system for the other engine(s).

(3) The minimum acceptable usable oil capacity, in terms of rotorcraft endurance and engine maximum oil consumption, is specified.

(4) The oil cooling provisions should be capable of maintaining the oil inlet temperature at or below the maximum allowable value.

b. Procedures.

(1) The requirement for an independent oil system for each engine should ensure continued adequate lubrication of each engine in the event of failure of the opposite engine(s) or of that opposite engine's oil system. The provision does not require that the engine oil system be independent of other components; e.g., the use of the engine's oil system for rotor drive system component lubrication is not precluded by this regulation.

(2) The usable oil capacity for each engine's oil system should not be less than the product of the maximum endurance of the rotorcraft times the engine's maximum oil consumption, plus some margin to ensure adequate circulation and cooling.

(3) Instead of a rational analysis of rotorcraft endurance and engine oil consumption rate, a usable oil capacity of 1 gallon for each 40 gallons of usable fuel may be used. (This concept should apply only to reciprocating engines.)

(4) Flight tests should be required to show adequate oil cooling provisions (reference § 27.1041).

AC 27.1013. § 27.1013 (Amendment 27-9) OIL TANKS.

a. Explanation. This regulation, along with § 27.1015, defines the oil tank design and installation requirements.

(1) The oil tank should be designed and installed to withstand, without failure, any vibration, inertia, fluid, and structural loads expected in operation.

(2) For reciprocating engines, the expansion space should not be less than 0.5 gallons or 10 percent of the tank capacity, whichever is greater.

(3) For turbine engines, the expansion space should not be less than 10 percent of the tank capacity.

(4) It should not be possible to inadvertently fill the expansion space with the rotorcraft in the normal ground attitude.

(5) Adequate venting should be provided.

(6) Oil overflow from the filler opening into the oil tank compartment should be prevented.

b. Procedures.

(1) The structural analysis of the tank, including the attachments, should ensure that the tank will not leak under the vibration, inertia, fluid, and structural loads expected in service.

(2) The expansion space may be determined by calculating the difference between the volume up to the vent opening and the volume to the spillover level of the filler opening. The expansion space volume must not be less than 10 percent of the volume to the filler spillover level (tank capacity) or not less than 0.5 gallons for reciprocating engine installations with oil tank capacities of 5 gallons or less.

(3) To assure adequate venting under all normal flight conditions, the tank should be vented from the top portion. Traps where condensed water vapor might freeze and obstruct the vent line should be avoided. If other components, perhaps an engine speed reduction gearbox, are vented to the engine oil tank, the oil tank vent line should be sized to handle this additional requirement as well as the air normally entrained in the return oilflow from the engine.

(4) A suitable method to prevent oil spillover from the filler opening from entering the compartment containing the oil tank would be a scupper with an attached drain line that discharges clear of the rotorcraft.

AC 27.1015. § 27.1015 (Amendment 27-9) OIL TANK TESTS.

a. Explanation. This regulation specifies the requirements which the oil tank tests should verify. Each oil tank should withstand, without leakage, an internal pressure of 5 PSI. This regulation also specifies that each pressurized oil tank used with a turbine engine must withstand, without leakage, an internal pressure of 5 PSI plus the maximum operating pressure of the tank.

b. Procedures. Test procedures for demonstrating these requirements are relatively simple and straightforward. Suitable adapters are fabricated to seal the various tank openings and also a fitting to introduce pressurized air into the tank. The air source needs to be regulated, and a suitable pressure gauge with a current calibration is required. Appropriate methods to check for leakage should also be available. This leak check can be a dip tank, soap and water mixture, or any other method which will provide acceptable results. Test fluid conforming to Federal Specification TT-S-735, Type III, or equivalent, is also an acceptable leak check substance.

AC 27.1017. § 27.1017 OIL LINES AND FITTINGS.

a. Explanation. This regulation outlines the certification requirements for oil lines and fittings.

b. Procedures.

(1) The line should be supported to prevent excessive vibration, and flexibility should be provided between points of relative motion. Advisory Circular 43.13-1A, Chapter 14, Section 2, Paragraph 709, may be used as guidance for the system design.

(2) Flexible hose must be approved. Generally, hoses listed in TSO-C53a or those qualified to equivalent military standards are accepted.

(3) The engine inlet and outlet oil lines should not have an inside diameter less than the corresponding inside diameter of the engine connection, and no line splices are permitted between connections; however, larger lines may be needed to ensure adequate oil flow to the engine or the transmission. Oils which exhibit high viscosity, long oil lines, and arrangements with little or no elevation of the tank outlet with respect to the engine inlet, are design characteristics which should be carefully checked.

AC 27.1019. § 27.1019 (Amendment 27-9) OIL STRAINER OR FILTER.

a. Explanation. This regulation defines the requirements for the engine oil system strainer or filter. If a strainer or filter which meets the requirements of this paragraph is incorporated as part of the type certificated engine, an additional airframe filter is not required.

b. Procedures. This paragraph requires an oil strainer or filter through which all of the oil flows for each turbine engine installation. The strainer or filter should be sized to allow oil flow at the flow rates and within the pressure limits as specified in the engine requirements. The effect of oil at the minimum temperature for which certification is sought should be accounted for.

(1) For each oil strainer or filter required by § 27.1019(a) which has a bypass, the bypass should be sized to allow oil flow at the normal rate through the oil system with the filtration means completely blocked.

(2) For each oil strainer or filter installed per this rule, the capacity must be such that when operating with oil contaminated to a degree greater than established during engine certification, the oil flow and pressure are within the operating limits established for the engine. The mesh requirements are determined by the engine installation documents for the filtration of particle size and density.

(3) Unless the filter is located at the oil tank outlet, § 27.1019(a)(3) requires an indicator that will show when the contaminant level of the filtration system, as specified in § 27.1019(a)(2), has been reached. The indicator should signal a contaminant level which will allow completion of the flight before the filter would enter a bypass condition. The indicator may be a pop-out button or other maintenance cue that is checked on each preflight.

(4) An evaluation of the construction and location of the bypass associated with the strainer or filter should be accomplished. The appropriate installation of the filter based on this evaluation would preclude the release of the collected contaminants in the bypass oil flow.

(5) If an oil strainer or filter installed in compliance with this regulation does not have a bypass, there must be a means to connect it to the warning system required in § 27.1305(r). This warning should indicate to the pilot the contamination before it reaches the capacity established in § 27.1019(a)(2).

(6) Section 27.1019(b) covers the blocked oil filter requirements associated with reciprocating engine installations. The lubrication system should be such that the normal oil flow will occur with the filter completely blocked.

AC 27.1019A. § 27.1019 (Amendment 27-23) OIL STRAINER OR FILTER.

a. Explanation. Amendment 27-23 relaxed an unduly restrictive requirement for an “indicator” to indicate the contamination level of oil filters. The rule change allows acceptance of a “means to indicate” the contaminate level to allow a wider range of acceptable methods of compliance.

b. Procedures. All of the policy material pertaining to this section remains in effect except that an “indicator” is not required to indicate the contamination level of the oil filters. Unless the filter is located at the oil tank outlet, § 27.1019(a)(3) requires that the oil strainer or filter have the means to indicate when the contaminant level of the filtration system, as specified in § 27.1019(a)(2), has been reached. If an indicator is installed, it should signal a contaminant level that will allow completion of the flight before the filter reaches a bypass condition. The indicator may be a pop-out button or other maintenance cue that is checked on each preflight inspection.

AC 27.1021. § 27.1021 OIL SYSTEM DRAINS.

a. Explanation. This regulation requires provisions be provided for safe drainage of the entire oil system with the rotorcraft at normal ground attitude and defines certain requirements for assuring that no inadvertent oil flow occurs from the system provided.

b. Procedures.

(1) The design of the oil system must provide a means for safe drainage of the entire oil system. This may require one or more drains depending on the design of the system. The routing of fluid lines should be such that drooping lines and fluid traps which are undrainable are avoided.

(2) The drain(s) must provide a means for a positive lock in the closed position. The method by which the lock is accomplished may be manual or automatic.

AC 27.1027. § 27.1027 (Amendment 27-23) TRANSMISSION AND GEARBOXES: GENERAL.

a. Explanation. Amendment 27–23 adds a new § 27.1027. This new section provides the regulations for rotorcraft transmission and gearbox lubrication systems. It incorporates lubrication system requirements that were derived from existing engine oil system requirements. These additional requirements have been adjusted or modified to reflect the needs of transmissions and gearboxes. Transmission and gearbox lubrication system regulations are similar to those for engines; therefore, reference is made to the engine lubrication sections as applicable.

b. Procedures.

(1) The pressure lubrication systems for rotorcraft transmissions and gearboxes should comply with the same requirements as the engine lubrication systems stipulated in §§ 27.1013 (except §§ 27.1013(c)), 27.1015, 27.1017, 27.1021, and 27.1337(d). These sections provide the requirements for oil tanks, tank tests, oil lines and fittings, and oil system drains.

(2) Each pressure lubrication system for rotorcraft transmissions and gearboxes should have an oil strainer or filter. The strainer or filter should:

(i) Remove any contaminants from the lubricant which may damage the transmission, gearbox, or other drive system component and any contaminants that may impede the lubricant flow to a hazardous degree.

(ii) Be equipped with a means to indicate that the bypass system (required by § 27.1027(b)) is at the point of opening due to the collection of contaminants on the strainer or filter, and;

(iii) Be equipped with a bypass system that will permit lubricant to continue to flow at the normal rate if the strainer or filter is completely blocked. In addition, the bypass system should be designed so that contaminants that have collected on the filter will not enter the bypass flow path when the system is in the bypass mode.

(3) Section 27.1027(c) requires a screen at the outlet of each lubricant tank or sump that supplies lubrication to rotor drive systems and rotor drive system components. The screen should remove any object that might obstruct the flow of lubricant to the filter required by § 27.1027(b). The requirements of § 27.1027(b) do not apply to the tank outlet screen.

(4) Splash-type lubrication systems for rotor drive system gearboxes should comply with §§ 27.1021 and 27.1337(d).

SUBPART E - POWERPLANT

COOLING

AC 27.1041. § 27.1041 (Amendment 27-2) COOLING--GENERAL.

a. Explanation. The rotorcraft design should provide for cooling to maintain the temperature of all powerplant and power transmission components and fluids within the limitations established for the items. Cooling provisions should be adequate for shutdown and for water, ground, and flight operating conditions. The adequacy of the cooling provisions should be demonstrated by flight testing.

b. Procedures.

(1) Test conditions and procedures necessary to demonstrate adequate cooling for water, ground, flight, and shutdown conditions should be agreed upon between the applicant and the FAA/AUTHORITY certification engineer. A cooling test proposal which defines the agreed test points and procedures should be prepared well in advance of the official certification testing.

(2) The test conditions selected would typically include climb, cruise, hover, and shutdown after a prolonged hover. Hover OGE should be evaluated if sling load operation is envisioned for the rotorcraft. One test condition which should be examined, particularly with regard to transmission cooling, is the point of highest multiengine mechanical power at the maximum ambient temperature. This is identified as test point “A” in figure AC 27.1041-1. The selection of test points should be tempered with engineering judgment and based on results from similar aircraft if such data are available. In showing compliance with the cooling requirements, the applicant should not be required to exceed rotorcraft established limits (gross weight, drive system torque, measured gas temperature, etc.), aircraft power required, or power available. The applicant may elect, however, to exceed these limits in order to minimize test points by conservative testing, or to anticipate future growth (increased gross weight, etc.).

(3) The need for a comprehensive cooling test plan prior to certification testing cannot be overemphasized. Highly derated engine installations, the relationship of power required to power available, the use of bleed air devices which would increase the measured gas temperature while aircraft power required remains the same, auxiliary cooling provisions, and the increase in engine temperatures with engine deterioration are factors which could affect the selection of cooling demonstration test points.

MULTIENGINE POWER AVAILABLE

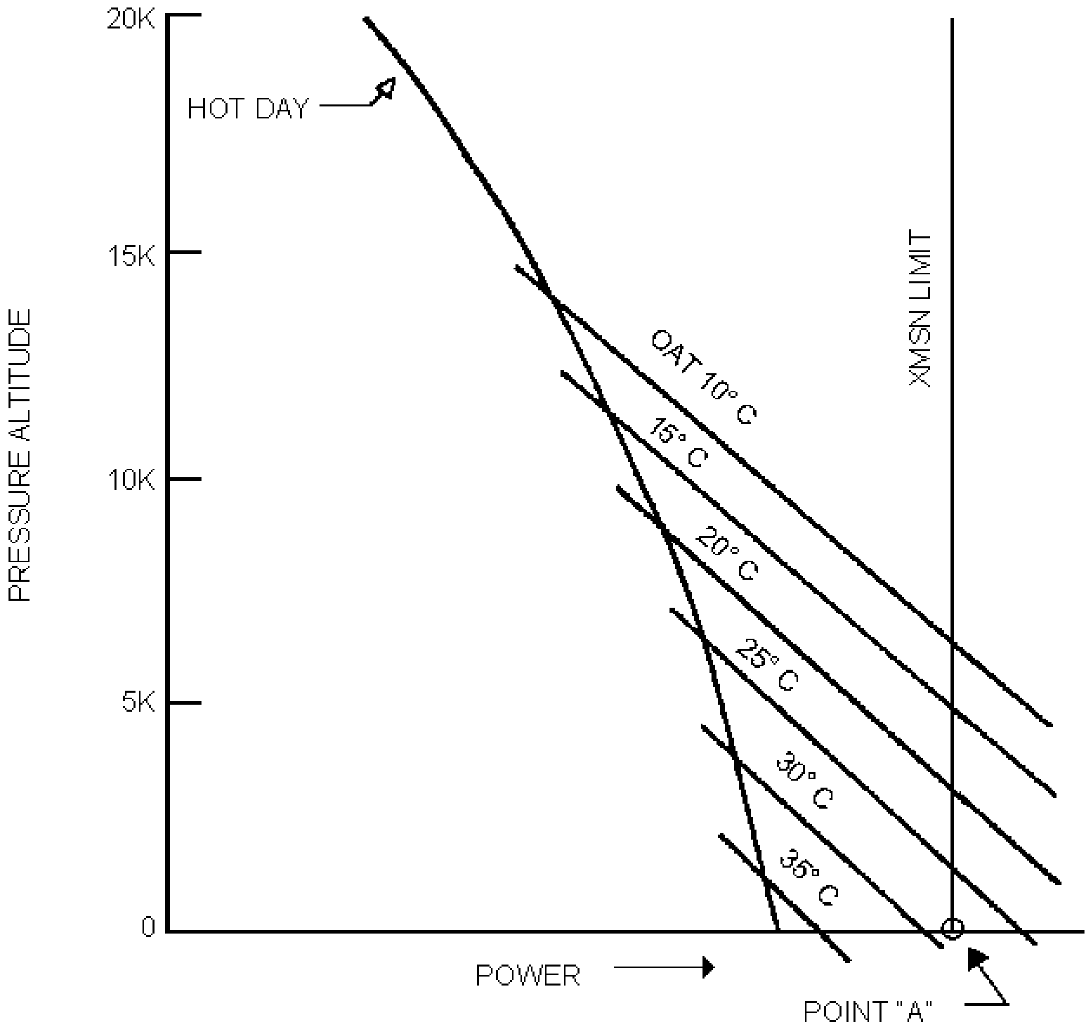


FIGURE AC 27.1041-1 ADDITIONAL COOLING TEST POINT

AC 27.1041A. <u>§ 27.1041 (Amendment 27-23) GENERAL</u>.

a. <u>Explanation</u>. Amendment 27-23 provides clarification and definition of powerplant components required to be considered when evaluating the performance of the powerplant cooling systems and arrangements.

b. <u>Procedures</u>. The rotorcraft design should provide for cooling to maintain the temperature of all powerplant and power transmission components and fluids within the limitations established for the items. Components to be considered include, but are not limited to, engines, rotor drive system components, auxiliary power units, and the cooling or lubricating fluids used with these components.

AC 27.1043. <u>§ 27.1043 (Amendment 27-14) COOLING TESTS</u>.

a. <u>Explanation</u>.

(1) This section defines the requirements for accomplishing the required cooling tests. Section 27.1043(a)(1) requires that certain ambient temperature correction factors be applied unless testing is accomplished at the maximum ambient atmospheric temperature prescribed. No corrected temperature may exceed specified limits. The requirement in § 27.1043(a)(4) that test procedures be in accordance with § 27.1045 does not limit testing to the conditions prescribed in that section. Section 27.1041(a) provides the basis for examination of other possible critical operating and shutdown conditions.

(2) This section establishes the hot-day condition as 100° F at sea level, decreasing 3.6° F per 1,000 feet of altitude above sea level. The applicant may select a lower maximum ambient atmospheric temperature for winterization installations. If the cooling tests are conducted under conditions deviating from the maximum anticipated air temperature, then the following correction factors are required unless another FAA/AUTHORITY-approved method is applicable.

(3) The temperature of engine fluids and powerplant components (except cylinder barrels) which have established limits must be corrected by adding to them the difference between the maximum anticipated air temperature and the ambient air temperature at the time of the first occurrence of the maximum component or fluid temperatures recorded during the cooling tests.

(4) Cylinder barrel temperatures must be corrected by adding 0.7 of the difference between the maximum anticipated air temperature and the ambient air temperature at the time of the first account of the maximum cylinder barrel temperature recorded during the cooling tests.

(5) During the cooling tests for reciprocating engines, the fuel used must be of the minimum grade approved for the engine and the mixture settings should be those

normally used in the flight stage for which the cooling tests are conducted. The carburetor parts list used during these tests becomes a requirement in the definition of the engine/carburetor configuration.

b. Procedures.

(1) Seldom is testing actually accomplished at the maximum required ambient temperature of at least 100° F at sea level lapsed 3.6° F per 1,000 feet pressure altitude. Component and fluid temperatures must therefore be corrected to derive the item temperature that would have been reached if the test day had matched exactly the maximum ambient temperature day. The applicant may select a higher maximum ambient temperature for cooling certification than the 100° F sea level hot day prescribed. Provisions are also made for selecting a maximum ambient temperature less than the 100° F sea level hot day for winterization installations not intended to function at the hot day conditions.

(2) When cooling test ambient conditions are cooler than the selected or prescribed hot day conditions, the applicant may take advantage of cooling air or fluid flows that would exist at hot day conditions. For example, thermostatically controlled oil cooler flow could be set for hot day conditions.

(3) The component and fluid temperature correction factor to be applied when test ambients do not correspond to the hot day conditions is commonly called the "degree-for-degree correction." It may be possible to justify, and the regulation allows the application of a more refined, less conservative correction factor. A correction factor other than degree-for-degree should be based on engineering test data.

(4) No corrected temperatures may exceed established limits. In order to maintain temperatures within established limits, the applicant may be willing to accept lesser performance than the full capability of a device. For example, a starter/generator capable of cooling under test cell conditions to 200 amperes continuous load may be limited to a lesser value, perhaps to 150 amperes, when installed in the aircraft due to cooling considerations. This continuous load for cooling must be equal to or greater than the allowable continuous load designated on aircraft instruments.

(5) If the engine or transmission cooling system heat load is increased in any way by rotorcraft configuration changes (affecting airflow, etc.), by other systems, by accessories (alternators, generators, etc.), or by any other heat source or potential heat source, then the maximum cumulative heat load from the worst-case combination of all these sources which is possible in service must be present during the cooling tests.

c. Thermal Limit Correction.

(1) An important correction factor which is not discussed in the regulations, but is frequently necessary to show the cooling adequacy required by § 27.1041, is the thermal limit correction factor. This factor is sometimes used if, at test day conditions,

the engine measured gas temperature does not correspond to that which would have occurred on a minimum specification engine at hot day conditions.

(2) The correction factor would not apply to those components not affected by changes in measured gas temperature (MGT) at a constant power. Typical items expected to be affected by changes in the MGT at constant power would be engine oil temperature, thermocouple harnesses, or other fluid, component, or ambient temperatures in the vicinity of the engine hot-section or exhaust gases. Other items remote from the hot-section, perhaps the starter-generator or fuel control, would not be expected to be influenced by MGT variations; however, the items affected and the magnitude of the factor to be applied should be established by testing.

(3) There are several acceptable methods for establishing the appropriate thermal limit correction factor during development testing. The general idea is to establish a stabilized flight condition, typically ground-run or IGE hover, and to vary the measured gas temperature at approximately fixed power and OAT conditions. This may be accomplished by utilizing engine anti-ice bleed air, customer bleed air, or by ingesting warmer than ambient air (either an external source or the engine bleed air) into the engine inlet. Care should be used in ingesting warmer than ambient air to ensure that the warm air is diffused in order to avoid possible engine surge.

(i) If it is not possible to attain a suitable variation in MGT by these methods, an acceptable, but more conservative, thermal limit correction may be obtained by allowing both shaft horsepower and MGT to vary at a stabilized flight condition and OAT.

(ii) The component temperature is plotted as a function of MGT, and the thermal limit correction from any test day MGT for any flight condition, to the MGT that would have existed with minimum specification engines on a hot day, is then applied to derive the final measured component temperature.

(4) In certain rare instances, it may not be required that the correction factor be applied to the full thermal limit capability of the engine. Consider the following example for the hot day hover IGE cooling test point at sea level.

	Power (SHP)	Corresponding MGT (°C)
Drive System Limit	900	---
Twin-Engine Hot Day Power Available	1050	750
Hot Day Power Required at Maximum G.W.	850	650
Engine Maximum Allowable MGT (Instrument Marking)	---	765
Test Day (90° F OAT) Parameters	850	600

(i) Notice that the installed hot day power available MGT from the engine performance program is 15° C cooler than the limit MGT (750 vs. 765° C), thus the engine has 15° C “field margin” which would allow the engine temperature to gradually increase 15° C to maintain a given power as engine life is utilized. Secondly, the measured gas temperature corresponding to hot day power required at maximum gross weight is less than that corresponding to either the drive system limit or twin-engine hot day power available. Thus, the thermal limit correction could be applied from the test day MGT, 600° C, to the power required MGT plus the field margin, 650° C plus 15° C, rather than applying the correction factor to the full thermal capability of the engine, 765° C.

(ii) Care should be used in applying this relieving method, because as the hover altitude changes, the maximum gross weight and power required (and the associated MGT) will vary. The data must be corrected to at least the maximum MGT for a minimum specification engine that can occur in service at the flight condition under investigation.

AC 27.1045. § 27.1045 COOLING TEST PROCEDURES.

a. Explanation.

(1) Section 27.1045(a) requires that cooling tests be conducted for the rotorcraft in the configurations and under the conditions most critical for cooling.

(2) Section 27.1045(b) requires that a temperature be stabilized prior to start of a cooling test for any test rotorcraft and any test stage. This is to ensure that the system reaches the maximum temperature from which it must be cooled. Temperature stabilization is achieved when the rate of change is less than 2° F per minute. Therefore, for each test rotorcraft in each test phase, the temperature must be stabilized prior to entry into flight test. If temperature stabilization cannot be achieved as a normal result of the entry condition, then operation through the full entry condition must be accomplished prior to entry into the flight test segment being conducted. This allows the temperatures to reach their maximum natural levels prior to test initiation. Also, for each test rotorcraft during the takeoff stage of flight, the climb at takeoff power must be preceded by a hover during which temperature stabilization is achieved.

(3) Section 27.1045(c) requires that a test must be conducted for each flight stage until either the temperatures stabilize, the flight test stage is completed, or an operating limitation is reached.

b. Procedures.

(1) To comply with § 27.1045(a), an applicant typically submits a cooling test proposal to the FAA/AUTHORITY for approval. The proposal should encompass detailed procedures to demonstrate cooling capability for each critical rotorcraft configuration and test condition (test point). If a single-most critical test configuration and condition is not readily identifiable (which is usually the case), then a series of cooling tests must be conducted. Typical cooling test segments are climb, takeoff and

climb, various cruise speeds and altitudes, hover, shutdown after prolonged hover, and sling load cooling if a sling is used. Any other appropriate test conditions and procedures necessary to demonstrate adequate cooling for water, ground, flight, emergency, and shutdown conditions should be addressed in the test proposal. For multiengine rotorcraft (particularly in regard to transmission cooling), one test point that should be investigated is the point of highest multiengine mechanical power at the maximum ambient temperature. Other significant test conditions to be considered for multiengine rotorcraft are the OEI test conditions. The selection of all test points should be tempered with engineering judgment and should consider test points and procedures used on previous, similar rotorcraft certification work, if available.

(2) Compliance with § 27.1045(b) is typically shown by use of existing cockpit instrumentation or add-on test instrumentation from which temperature data are read prior to and during test segments. Test plans should clearly identify what is to be used and who is authorized to make and record readings and the accuracy and current calibration requirements for test instrumentation.

(3) Compliance with § 27.1045(c) is typically shown during FAA/AUTHORITY authorized testing by conducting each test segment until at least one of the three criteria (temperature stabilization, flight test segment completion, or an operating limitation) is reached. If an adverse operating limitation is reached, such as overheating, or the test cannot otherwise be successfully completed, then compliance has not been shown and a reevaluation is required.

AC 27.1045A. <u>§ 27.1045 (Amendment 27-23) COOLING TEST PROCEDURES</u>.

a. <u>Explanation</u>. Amendment 27-23 clarifies acceptance criteria for the powerplant cooling tests that are appropriate if, during the cooling test, component temperatures peak and then decline rather than stabilize. In these instances, the previous requirement to continue the test until "stabilization" occurred was unduly restrictive and was eliminated.

b. <u>Procedures</u>. All of the policy material pertaining to this section remains in effect except that the engine fluid temperatures do not have to stabilize.
Paragraph AC 27.1045 currently lists three criteria for test completion: temperature stabilization, flight test segment completion, or an operation limitation. With Amendment 27-23, a fourth criteria for test completion is: 5 minutes after the peak temperature is reached, the test can be considered complete.

SUBPART E - POWERPLANT

INDUCTION SYSTEM

AC 27.1091. § 27.1091 (Amendment 27-2) AIR INDUCTION.

a. Explanation.

(1) The air induction system for each engine should be of a configuration to supply the air required under the operating conditions for which certification is required.

(2) The intake system shall be designed such that if a backfire flame occurs, it will emerge outside the engine compartment cowling.

(3) Where required in the induction system, drains must be provided which discharge clear of the rotorcraft and out of the path of exhaust flames.

(4) For rotorcraft powered by a turbine, the inlets should be located or protected to minimize foreign object ingestion as defined in the regulation. The inlets must be protected during takeoff, landing, and taxiing. There must also be means to prevent leakage of hazardous amounts of flammable fluids from entering the engine intake system.

b. Procedures.

(1) For turbine-engine installation, the induction system should supply air of suitable quality to meet the installation requirements of the engine manufacturers. The installation requirements should be met throughout the operating envelope of the rotorcraft. In addition, the design and location of the air induction system should prevent accumulations of rain or hail, either external or internal to the induction system, that could adversely affect engine operation.

(2) The inlet design should account for the prevention of hazardous fluids entering the engine. Some designs will have inlet ducts which are free from any fluid lines; however, other designs may route the engine inlet air through a compartment which has flammable fluid lines. When the condition exists, test demonstrations of critical leakage during operations have been used to substantiate the installation. The fluid leakage may not have an adverse effect on engine operation.

(3) The air induction system design should also account for and minimize the possibility of foreign matter ingestion during takeoff, landing, and taxiing.

(4) For reciprocating engine installations, the induction system should supply air of suitable quality and quantity to the carburetor inlet of the engine. The condition of the air at the entering face of the carburetor is extremely important. For proper

operation, it is essential that the airflow be smooth, uniform, clean, and unrestricted throughout the very wide range of horsepower expected from the engine.

AC 27.1091A. § 27.1091 (Amendment 27-23) AIR INDUCTION.

a. Explanation. Amendment 27-23 removed § 27.1091(d) since the specific test defined by this paragraph was not critical for certain rotorcraft. The turbine inlet foreign-object-ingestion protection provided by § 27.1091(d) is adequately evaluated by existing requirements in § 27.1091(e)(2).

b. Procedures. This rule change did not change the current suggested methods of compliance.

AC 27.1093. § 27.1093 (Amendment 27-20) INDUCTION SYSTEM ICING PROTECTION.

a. Reciprocating Engines.

(1) Explanation.

(i) Atmospheric moisture, even in clean air and temperatures above freezing, can result in ice accumulations in induction systems to a degree which can easily cause engine failure.

(A) Impact Ice. This forms as supercooled water droplets impact on engine induction system components. Particularly heavy accumulations must be expected where bends or turns in the induction system force changes in the airflow direction thus centrifuging the droplets out of the air stream where they freeze on impact with induction system components. A serious form of impact ice is the collection of ice on fuel metering elements of the carburetor, the alternate (preheat) valve, and any screens in the system.

(B) Throttle Ice. This type of ice forms at or near the throttle in a partly closed position (up to 30° F) due to cooling effect resulting from the increase in kinetic energy (increased velocity) of the air in the restricted flow area.

(C) Refrigeration Ice. This forms as a result of the cooling effect of the fuel evaporating after the fuel is introduced into the airstream. For some float type carburetors, it is possible in rare instances to accumulate serious ice during a closed throttle glide with ambient air temperatures as high as 93° F and relative humidity of 30 percent. At low cruise power, ice can occur at outside air temperatures as high as 62° F and relative humidities as low as 60 percent. Most of the heat necessary to evaporate fuel is supplied from the air as it drops in temperature. Fuel evaporation ice can affect airflow by blocking the throat of the manifold riser; it can affect the fuel-air-ratio by interfering with the fuel flow; and it can affect mixture distribution or quantity of mixture flowing to individual cylinders by upsetting the fuel flow distribution, or quantity of mixture flowing to individual cylinders by upsetting the fuel flow distribution at the fuel nozzle or airflow distribution in the manifold throat. This refrigeration phenomenon is the most serious of all factors causing carburetor ice.

(2) Procedures. Normally, flight tests with carburetor air temperature instrumentation are required. Unless otherwise justified, conduct all tests at maximum gross weight, a median center of gravity, in level flight, and at the engine speed which, considering cooling fan effects, if any, produces the minimum heat to the carburetor muff or engine component area utilized to provide the carburetor air heat. The optimum test condition is flight at the altitude at which the measured OAT is 30° F and a power setting of 75 percent maximum continuous power can be maintained. If this combination cannot be achieved, satisfactory interpolation of data from other test conditions can be achieved using the methodology of AC 23-8A, Flight Test Guide for Certification of Part 23 Airplanes.

(i) In addition to the preheat requirements of § 27.1093(a)(1), (a)(3), and (a)(4), the design should consider the possibility of impact ice (supercooled droplets below freezing) on engine air inlet components opening into the airstream. However, normal practice is to provide a crew selectable, sheltered, alternate engine air intake arrangement which, by inspection can be determined to be free from impact ice accumulations. Typically, a sheltered alternate air source would be acceptable if the opening is located inside the cowling out of the free airstream. However, precautions should be taken to ensure that backfire flames to be expected do not constitute a fire or explosion hazard.

(ii) For further information review AC 20-113, Pilot Precautions and Procedures to be Taken in Preventing Aircraft Reciprocating Engine Induction System and Fuel System Icing Problems, NACA TN 1790, NTSB AAS-72-1, DOT/FAA/CT 84/44 (1982) and NACA TN 1993 (1949).

b. Turbine Engines - Ice Protection.

(1) Explanation.

(i) This rule requires turbine engines and turbine-engine inlets to perform satisfactorily in atmospheric icing conditions defined in Appendix C of Part 25. On an equivalent safety basis, the limited icing envelopes described in paragraph AC 29.877 may be used to show compliance with the intent of the regulation if the rotorcraft is limited to not greater than a 10,000-foot pressure altitude for all operations. If operations are permitted above 10,000 feet, the Appendix C, Part 25, envelope must be used from 10,000 feet to the service ceiling or 22,000 feet. These possible equivalent safety approaches are not discussed herein. Compliance with the induction system icing protection rule is required regardless of flight manual limitations or restrictions against flight into atmospheric icing conditions.

(ii) In showing compliance with § 27.1093(b)(1)(i), the FAA/AUTHORITY has accepted the concept of limited exposure associated with escape from inadvertent ice encounters.

(A) Under the concept of limited exposure associated with escape from inadvertent ice encounters, it is presumed that there will be a flight manual limitation against flight into known icing, and that the engine induction system will be reevaluated if total aircraft ice protection certification is requested. Under this concept, the rotorcraft is assumed to fly directly through the icing environment; i.e., direct sequential penetration and straight line exit from both the continuous maximum and intermittent maximum icing clouds. Thus, the duration of exposure to the icing environment could be calculated by knowing the aircraft flight speed and cloud horizontal extent. A range of engine power and rotorcraft airspeeds should be evaluated to encompass the operating envelope of the rotorcraft. Note that aircraft speed has a pronounced effect (Ludlam effect) on ice accretion on small surface areas (inlet screens). A review of this phenomena may be found in National Research Council (Canada) Letter LT-92.

(B) When this limited exposure concept is used, the aircraft type certificate data sheet should clearly specify that the engine induction system must be reevaluated if certification to the general ice protection regulation, § 27.877 or § 27.1419, is requested. This direct penetration and exit approach is inappropriate for aircraft for which full icing clearance is requested (reference § 27.1419).

(iii) Engine induction system continuous icing protection would be necessary for aircraft for which full-icing clearance is requested (reference § 27.1419(d)). The approach is much preferred for all programs in order to reduce the scope of any eventual total aircraft icing program effort and to increase the safety level in conducting the rotorcraft natural icing tests. Since at least one rotorcraft has been FAA/AUTHORITY certificated to operate in known icing conditions and others have active development programs to this end, applicants should anticipate eventual full-icing clearance and consider that the engine induction system may be required to operate routinely in a continuous icing environment.

(iv) It is noted in paragraph AC 29.877 that some natural icing tests are required to show compliance with the overall rotorcraft ice protection requirements. It is not required that the engine induction system be evaluated as a part of that natural icing test if adequate verification has been shown by tunnel testing, analysis, or other means to ensure satisfactory operation in an extended continuous icing environment. If, however, subsequent rotorcraft natural icing testing shows unanticipated detrimental engine inlet effects, the inlet ice protection system should be reexamined.

(v) The regulation specifies the examination of flight idling conditions. This requirement is normally associated with a low-power letdown at the minimum practical forward airspeed. Alternatively, evaluation of the minimum power and minimum airspeed combination specified in the RFM for operation in visible moisture when below 40° F will accomplish the intent of the idling requirement.

(vi) An acceptable approach to a finding of compliance would be a combination of analysis of the performance of the ice protection system which covers the range of the applicable icing flight envelope (maximum altitude, minimum temperature, etc., of the basic rotorcraft) supported and validated by tests. Ideally, these tests would be conducted in natural atmospheric ice with special instrumentation for droplet size and liquid water content. In practice, however, natural icing testing may pose unacceptably severe problems since rotorcraft may not have the range and speed to reasonably find icing clouds and may not be equipped with the airframe and rotor ice protection needed for safety during the testing.

(vii) Problems with analysis emerge if engine inlets incorporate screens, turning vanes, sideward or upward openings, and edge or lip configurations which deviate from the airfoil shapes assumed in most of the analytical procedures described in current technical literature. The applicant should recognize that if meaningful analytical methods are not available, extensive testing with significant conservatism or possibly design changes may be required. Inlet screens in particular, if not adequately heated, fall in this category and can only be accepted if shown by very conservative ice testing to not significantly impede airflow to the engine.

(viii) The icing evaluation should definitely include some test points or other adequate evaluation of flight at ambient temperatures several degrees Fahrenheit above freezing and with very high water content. This condition has actually produced multiengine flameouts in in-service aircraft. The actual icing phenomena involved is not fully understood and in some instances efforts to duplicate the phenomena in icing tunnels were unsuccessful. Usually this condition does not produce rotor icing; therefore, actual flight testing using special precautions to ensure safe autorotation landings or engine relight capability may be needed to identify this condition.

(2) Procedures.

(i) Review paragraph AC 29.877, ADS-4, Report No. FAA-RD-77-767; Aircraft Icing Handbook, FAA Technical Report No. DOT/FAA/CT-88/8, and Advisory Circular 20-73, Aircraft Ice Protection. (The comparative concept described under Item 34 of AC 20-73 is obsolete and should not be considered.) These data provide extensive description and methodology for evaluation of ice protection systems; however, as noted above, these data generally apply to near straight line droplet trajectory with impingement onto conventional airfoil shaped inlets. As such, the applicability of these data to rotorcraft engine inlet ducts is limited and may require extensive adjustment to accommodate the different inflow trajectories and shapes of rotorcraft.

(ii) An analysis, appropriate to the configuration; i.e., heated or unheated impingement surfaces, should be prepared. To be acceptable, this analysis should show the inlet to be adequately protected by heat, or if unheated, to show that the inlet with ice accretions as predicted, will provide adequate airflow to the engine throughout the flight envelope of the rotorcraft.

(A) For heated surfaces, ADS-4 and Report No. FAA-RD-77-76 provide detailed suggestions on heat transfer analysis particularly applicable to bleed air heated inlet lips formed in airfoil shapes. These data are limited in applicability and may not be useful for analyzing engine inlet water droplet trajectories to be expected at low airspeed and high engine airflow. Actual icing tests may be needed to derive the impingement patterns for these conditions.

(1) Acceptability criteria for heated inlet ducts usually require sufficient heat to evaporate the water to be expected in a “continuous maximum” icing cloud and to anti-ice the duct during flight in “intermittent maximum” icing clouds, providing the run-back and refreeze to be expected does not cause additional airflow disruption or damage to the engine. Full-scale inlet icing tests with the engine installed and operating should be conducted to verify the analysis. Engine power changes which may be expected in service should be included in the testing. Wind tunnels equipped for icing tests probably are the most useful means of conducting these tests if natural icing tests are impractical. The rotor downwash effect should be considered to the extent possible by adjusting the inflow angle in the tunnel.

(2) The power loss (bleed air, generator load, etc.) attributable to the heating requirements will affect the performance of the rotorcraft. Normally, this may be accounted for by specifying a gross weight incremental deduction from the flight manual performance data for flight into visible moisture below 40° F.

(3) Special evaluation of the possibility of ice ingestion damage to the engine should be made for heated systems which considers the ice ingestion to be expected when the anti-ice system is actuated after a delay of 1 minute for the pilot to recognize that the rotorcraft has encountered ice. This time delay may be reduced if the crew is provided adequate distinctive cues to alert them that the rotorcraft has encountered icing conditions.

(B) For unheated inlets, an acceptable method for showing compliance would include an extensive, detailed analysis (which shows that ice accretions on and in the inlet do not seriously obstruct adequate airflow to the engine) and tests as necessary to validate the analysis. The analysis of ice accretion becomes even more questionable since the unheated inlet involves ice buildups which themselves progressively change shape during icing exposure.

(1) Flight testing with an instrumented rotorcraft in natural ice to verify the analysis is desirable; however, wind tunnel tests as discussed above may be used. Since unheated inlets typically continue to accrete ice as a function of exposure, both the analysis and the test should realistically consider the actual exposure to be expected in service. This should not be less than penetration of the continuous maximum icing cloud followed immediately by exposure to the intermittent maximum

cloud for rotorcraft not certified for icing. Engine power changes which may be expected in service should be included in the testing, and a warm-up period at the conclusion of the icing exposure should be shown for some selected test points to evaluate potential ice breakaway and ingestion.

(2) For the nonicing certified rotorcraft using the limited icing exposure concept for inlet certification, some conservatism should be applied to account for the fact that inlet icing may occur without airframe icing, and that the escape procedure from this unapproved operating condition is not defined. A demonstration of 30-minute hold capability in the continuous maximum cloud would be acceptable. Alternatively, if positive cues (perhaps a carefully located ice detector) of potential inlet icing are provided to the crew, the time increment could be reduced to recognition plus 15 minutes (15-minute escape time after recognition is consistent with the single ice protection system failure recognition and escape guidance for aircraft ice protection systems in paragraph AC 29.877). It should not be assumed that airframe icing will always be available as a cue to potential inlet icing. The main rotor, for example, may not show icing indications above 25° F, whereas some inlets may ice critically near 32° F ambient. A reduction of the acceptable 30-minute exposure should not be based on observation of ice accretions on protruding components which are likely to be changed. For example, a limited exposure inlet icing program which reduces the inlet icing exposure time based on crew recognition of icing on the windshield wipers may be invalidated at a later date if a new windscreen deletes the wipers.

(iii) Inlet capability during IGE hover in icing conditions has not generally been considered for rotorcraft not certified for icing. However, the FAA/AUTHORITY is recently aware that some inlets may ice at zero airspeed near 32° F with no indications of airframe icing in the field of view of the crew. This special concern of operating within RFM limitations, and yet placing the induction system in jeopardy, may be addressed in several ways. If the induction system ice protection scheme is not dependent on airspeed for proper function, the issue may be addressed by tunnel testing with inlet airflows approximating hover with no particular attention to tunnel windspeed. For protection schemes which may be sensitive to airspeed (external screens have shown this tendency), actual hover demonstration at or near zero speed tunnel conditions may be appropriate. Icing detectors located to indicate induction system icing in hover may be an option to a hover icing protection demonstration. Recently, on an external screened configuration, the FAA/AUTHORITY has accepted a satisfactory IGE hover demonstration of 30 minutes at the critical ambient temperature (i.e., ambient consistent with no airframe icing but potential inlet icing), 0.6 grams/meter^3 LWC, and 40-micron droplet size as an adequate response to this concern.

(iv) For aircraft requesting full icing approval, or for those electing to show continuous induction system icing protection, the forward flight icing exposure would not be less than that time required to stabilize any ice accretions observed during repeated cycles of the continuous maximum followed by intermittent maximum cloud exposure. Typically, any ice accretions resulting from these repeated cycles would be expected to stabilize in less than 30 minutes. The 30-minute hold capability in the continuous maximum icing environment could thus be ensured without special testing by careful selection of the test points for this repeated cycle.

(v) A rotorcraft requesting full icing approval should also have hover capability in the icing environment. Intermittent maximum icing conditions are not likely to exist near ground level and a satisfactory demonstration could involve the ability to hover indefinitely in the continuous maximum icing environment. Alternatively, carefully worded RFM limitations to restrict hover time may be acceptable if the system is not capable of indefinite exposure. Hover capability verification may not involve zero airspeed demonstration if the inlet protection system is insensitive to rotorcraft airspeed.

(vi) The engine(s) must be installed or protected to avoid engine damage from ice ingestion due to ice accretion in the inlet or on other parts of the rotorcraft, including the rotors, which may break away to enter the inlet. If screens or bypass arrangements are provided for these purposes, they should be included in the icing tests and shown by test or rational analysis to effectively protect the engine.

(vii) For unheated inlets, significant ice accumulations to be expected on the inlet may adversely affect the engine stall margin, acceleration characteristics, duct loss, etc. Dry air flight tests to evaluate these aspects can be accomplished by affixing ice shapes to the inlet. These shapes should closely match the actual ice shapes defined by test or analysis. In addition, it should be determined that ice shedding into the engine inlet either during continued flight into icing conditions or after emerging from the icing environment does not damage engine compressor on other inlet components.

c. Turbine Engines - Snow Protection.

(1) Explanation.

(i) Section 27.1093(b)(1)(ii) provides that the turbine engine and its air inlet system operate satisfactorily within the limitations established for the rotorcraft, in both falling and blowing snow. The section does not provide the definition of falling and blowing snow.

(ii) Since the regulation provides for certification “within the limitations established for the rotorcraft,” the FAA/AUTHORITY can accept a restriction against snow operations in the limitations section of the RFM in lieu of demonstration of compliance. If no restriction on snow operations appears in the RFM, it is presumed that the aircraft may operate in snow at the pilot’s discretion.

(2) Acceptance Criteria.

(i) The FAA/AUTHORITY has accepted that engine induction system operation in falling and blowing snow can be approved without restriction if normal operations under the following conditions are demonstrated:

Visibility: one-quarter mile or less as limited by snow.

Temperature: 25° F to 34° F (28° F to 34° F desired), unless other temperatures are deemed critical.

Operations: Ground operations - 20 minutes
IGE hover - 5 minutes
Level flight - 1 hour
Descent and landing

(ii) RFM visibility restrictions for falling and blowing snow operations are not appropriate.

(iii) Time limitations, other than possibly for ground and hover operations, are not appropriate.

(iv) Artificially produced snow should not be used as the sole means of showing compliance.

(3) Rationale.

(i) The test conditions specified--visibility, temperature, and operations--are based on previous certification programs, previous FAA/AUTHORITY guidance, and on research by the FAA technical center and others.

(A) Visibility. The test visibility defined, 1/4-mile visibility or less as limited by snow, represents a heavy snowstorm and is the maximum likely to be encountered in service. Rotorcraft which have been certified to the 1/4-mile visibility test criteria have not shown engine inlet snow-related service difficulties. It is important to note that the visibility specified is a test parameter rather than an operational limitation to be imposed on the rotorcraft after the tests are completed.

(B) Temperature.

(1) The ambient temperature specified is conducive to wet snow conditions. Wet snow tends to accumulate on unheated surfaces subject to impingement.

(2) Colder ambients, more conducive to dry snow conditions, may be critical for some induction systems. Colder exterior surfaces may be bypassed, and the snow crystals may stick to partially heated interior surfaces where partial melting and refreezing may occur.

(3) Company development testing or experience with very similar type induction systems may be adequate to determine the critical ambient conditions for certification testing.

(C) Operations.

(1) Ground running, taxiing, and IGE hover operations are generally the most critical since the rotorcraft may be operating in recirculating snow. Twenty-five minutes under these extreme conditions would seem a reasonable maximum, both from the view of pilot stress and the maximum expected taxi time prior to takeoff in bad weather.

(2) One hour of level flight operation under ¼-mile visibility snow conditions should provide ample opportunity for hazardous accumulations to begin to build.

(3) The descent and landing will provide an engine power change, an induction system airflow change, and a variation in the external airflow pattern near the induction system entrance. The initiation of the descent and final flare for landing may also produce additional airframe vibration transmitted to the induction system. These power, airflow, and vibration changes may provide an opportunity for any level flight accumulations to be ingested into the engine. Hazardous accumulations are not acceptable during or after any test phase.

(ii) Visibility may fluctuate rapidly in snowstorms. It is affected by the presence of fog or ice crystals, is not crew measured or controlled, and is difficult to estimate. A visibility operational limitation based on snow, therefore, is not appropriate.

(iii) Since during cruise in snow conditions the aircraft is likely to be in and out of heavy snowfall, it is not practical for the crew to account for the time spent in snow in level flight conditions. Thus, it is not appropriate to include time limitations in the RFM for level flight snow operations.

(iv) A practical ground and IGE hover time limitation of less than 25 minutes in recirculating snow may be considered. The expected action at the expiration of this specified time period would be shut down and inspection of the inlet system or transition to a safe flight condition where demonstration has shown that moisture accumulations will not intensify or shed and cause engine operational problems.

(v) Artificially produced snow is an excellent development tool and has been successfully used to indicate potential problem areas in induction systems. These devices are usually restricted to use for hover and ground evaluations, and the snow pellets produced by these machines are not sufficiently similar to natural snowflakes to justify the use of artificial snow as the sole basis of certification.

(4) Procedures.

(i) Satisfactory demonstration of the test conditions requires that the engine, induction system, and proximate cowling surfaces remain free of excessive snow, ice, or water accumulation. Excessive accumulation is defined as accumulation that may cause engine instability, damage, or significant loss of engine power. If a questionable amount of snow or moisture accumulates in the inlet, the applicant may elect to demonstrate that this amount in the form of snow or water and ice, as appropriate, can be ingested by the engine without incurring surge, flameout, or damage.

(ii) The conditions specified assume actual flight demonstration in natural snow. The ground operations and IGE hover test conditions assume operation in recirculating snow. Blowing snow, resulting from rotor airflow recirculation, can be expected to be more severe than natural blowing snow if the rotorcraft continues to move slowly over freshly fallen snow. Thus, the blowing snow operational capability is usually demonstrated by the taxi and hover operations in recirculating snow.

(iii) For VFR rotorcraft, the airspeeds for the level flight test condition should include the maximum consistent with the visibility conditions. For IFR operations, the airspeed should be the maximum cruise speed or the maximum speed specified for snow operations in the flight manual limitations, unless other airspeeds are deemed more critical. It is recognized that many rotorcraft initially VFR certified are later IFR certified with a resulting possible increase in airspeed in snow conditions. This factor should be considered if IFR certification is anticipated.

(iv) The visibility specified assumes that visual measurements are made in falling snow in the absence of fog or recirculating snow by an observer at the test site outside the tests rotorcraft's area of influence. An accepted equation for relating this measured visibility to snow concentration is $V = 374.9/C^{0.7734}$ where C is the snow concentration (grams/meter^3) and V is the visibility (meters).

(A) This equation can be reasonably applied to all snowflake type classifications and is credited to J.R. Stallabrass, National Research Council of Canada.

(B) Other equations may be applied if they are shown to be accurate for the particular snowflake types for the test program.

(v) The snow concentration corresponding to the 1/4-mile or less visibility prescribed will be extremely difficult to locate in nature. Data from Ottawa, Canada, research indicate that fewer than 4 percent of the snowstorms encountered there meet the 0.91 grams/m^3 concentration associated with the 1/4-mile visibility. Furthermore, the likelihood that the desired concentration will exist for the duration of the testing is even more remote. Because of these testing realities, it is very likely that exact target test conditions will not be achieved. Those involved in certification must exercise good judgment in accepting alternate approaches.

(vi) For some engine induction systems, it may become apparent by inspecting for moisture accumulations that ground and IGE hover operations in recirculating snow are much more severe than the level flight test. In this instance, it is reasonable to accept prolonged IGE operations in recirculating snow and to accept durations of less than 1-hour level flight in 1/4-mile or less visibility. Best efforts should be made to ensure that at least some level flight time is accomplished at 1/4-mile or less visibility to ensure that the spectrum is covered.

(vii) It should be determined that the visibility established at the test sight is limited by snow and not by fog or poor lighting (twilight) conditions.

(viii) The concentration of snow approaching the inlet in severe recirculation will far exceed the quantity encountered in the natural snowfall. Recirculation is necessarily a qualitative judgment by the test pilot. The snow concentration at the inlets during recirculation would vary for different rotorcraft types and would be dependent on rotor characteristics, power setting, and inlet location. For test purposes, recirculation should be the highest snow concentration attainable in the maneuver, or that corresponding to the lowest visibility at which (in the pilot's judgment) control of the rotorcraft is possible in the IGE condition. The 1/4-mile or less visibility specification outside of the recirculation influence becomes inconsequential provided that fresh, loose snow is continually experienced during the ground operation and IGE hover testing phase. However, since it is intended that the test phases be accomplished sequentially to ensure that transition to takeoff and other transients are considered, the conditions at takeoff, level flight, and descent and landing should approximate the 1/4-mile visibility criteria.

d. Turbine Engines - Ground Icing.

(1) Explanation. This requirement addresses the situation where extended ground operation in icing exposes the rotorcraft and its engine inlet to icing (ground fog) conditions which may have different droplet impingement patterns and involve different and/or less effective means of ice protection. Note that the requirement is effective in Amendment 10 and is applicable regardless of any desire to prohibit dispatch into known icing conditions.

(2) Procedure. Since this condition assumes zero airspeed, wind tunnel testing may be inappropriate unless conservative extrapolation of low speed tunnel data can be determined to be valid. For protection schemes which are dependent primarily on airspeed for proper functions (external screens have shown this tendency), it may be necessary to verify adequate ground operation protection capability by very low speed tunnels or by the use of outside facilities such as the Canadian National Research Council's spray rig at Ottawa, Canada. For heated systems or for internal bypass schemes, tunnel speed may not be important, and adequate demonstration may be accomplished at higher tunnel speeds provided that internal inlet airflows and heat available are properly considered. Testing should proximate the regulatory test conditions and be continued for 30 minutes using engine power and control

manipulation as normally accepted during taxiway operations, followed by an acceleration to takeoff power. The test time may be shortened if deice/anti-ice protection is adequate or if stabilization of ice build-up is affirmed. The induction system should be in condition for safe flight at the conclusion of the test.

e. § 27.1093(c) Supercharged Reciprocating Engines.

(1) Explanation. This rule authorizes the designer to take credit for the heat-of-compression available downstream of an engine air inlet supercharger to meet the induction system heat rise requirements of § 27.1093(a)(3) or (a)(4), provided the heat rise is automatically available for the applicable altitude and operating condition.

(2) Procedures. Since a wide variety of superchargers and supercharger controls (waste-gate controls) have been devised, it is impracticable to outline specific instructions for determining compliance. However, the certification engineer can properly evaluate the arrangement by analyzing the system for trends (in heat rise available) and conducting measurements to verify these trends and quantify the actual values of heat rise available. Some factors to be considered are:

(i) Mechanically driven superchargers for rotorcraft usually operate over a very narrow speed range, thus the heat of compression (Δ temperature) may remain constant over the altitude range. Conversely, turbosuperchargers usually are controlled (via waste-gate position modulation) to gradually increase the compression with increase in altitude, thus the critical (or lowest) Δ temperature may be available at very low altitudes.

(ii) Other waste-gate controllers sense carburetor deck pressure and respond by modulating the waste gate to maintain a constant carburetor deck pressure within the capabilities of the turbo unit. Heat of compression may then vary with altitude and engine power in a complicated fashion such as to require experimental temperature measurements across a wide range of operating conditions to determine compliance.

(iii) Turbosuperchargers which are not controlled (by waste-gate modulation) but respond to an orificed exhaust generally will (at constant power) produce more heat rise at altitude than at sea level; however, size matching between engine and turbo unit may affect this. Instrumented flight tests should be used as a final compliance verification method.

AC 27.1093A. § 27.1093 (Amendment 27-23) INDUCTION SYSTEM ICING PROTECTION.

a. Explanation. Amendment 27-23 clarifies that the phrase, “within the limitations established for the rotorcraft” applies only to the requirement in § 27.1093(b)(1)(ii) for demonstrating flight in falling and blowing snow.

b. Procedures. All of the policy material for this section remains in effect with the update that turbine engines and turbine engine inlets should perform satisfactorily in atmospheric icing conditions defined in Appendix C of FAR 29 instead of FAR 25. In addition to the procedures of paragraph AC 27.1093, the following procedures should be followed:

(1) A “serious loss of power” in this section has been interpreted to be any power loss that requires immediate pilot action. In addition, the term “adverse effect on engine operation” in § 27.1093(b)(1)(ii) has been interpreted to be an effect that would prevent the engine from achieving rated aircraft flight manual performance (takeoff/climb/etc.). This term also includes effects on the engine induction system characteristics to an acceptable level established by the engine manufacturer (inlet distortion, etc.).

(2) It should be shown that rotorcraft that are prohibited from flight into falling and blowing snow can exit inadvertent entrance into those conditions without adverse effect upon the operating characteristics of the engine or the rotorcraft.

(3) For full flight capability into snow, both falling and blowing, it should be shown that each engine, and its inlet system, will operate satisfactorily throughout the flight power range of the engine and the operating limitations of the rotorcraft. It should be shown that any build-up or accumulation of snow will not reduce or block the flow of inlet air to the engine. Any accumulations that become dislodged should not affect engine operation.

SUBPART E - POWERPLANT

EXHAUST SYSTEM

AC 27.1121. § 27.1121 (Amendment 27-12) EXHAUST SYSTEM--GENERAL.

a. Explanation.

(1) This section addresses the arrangement of exhaust components and the protection against hazardous conditions which exist with hot exhaust gases.

(2) The objective is to allow for thermal expansion of manifolds and pipes, prevent local hot spots, and eliminate the possibility of igniting flammable fluids or vapors.

b. Procedures.

(1) Sufficient clearance of hot exhaust components must be maintained from structure, fuel cells, flammable fluid lines, and electrical components to compensate for thermal growth under normal and most extreme operating temperatures. Verify that adequate clearance exists between the exhaust system components and the surrounding structure, and that no interference occurs under the most adverse temperature excursions.

(2) Hot spots that can occur on fuselage or rotor blade skin as a result of impingement or in compartments due to an accumulation of hot gases should be eliminated with deflectors or by providing adequate flow-through ventilation. Compliance may be shown by demonstration or analysis.

(3) It should not be possible to ingest sufficient quantities of exhaust gases which will produce engine surges, stalls, or flameouts during normal and emergency operation within the range of operating limitations of the aircraft and of the engine. Analysis and/or flight testing may be required to demonstrate compliance. If flight testing is required, particular attention should be placed upon critical azimuths and wind conditions.

(4) Exhaust system surfaces hot enough to ignite flammable fluids or vapors must meet the isolation or shielding requirements of this section in addition to the requirements of §§ 27.1183 and 27.1185. Good design practice suggests that the isolation and shielding features incorporated would continue to be effective under the emergency landing conditions specified in § 27.561.

(5) It should be demonstrated that exhaust gases are discharged in such a manner that they do not cause distortion or glare which seriously affects pilot visibility at

night. One method of compliance would be a night flight evaluation at critical azimuths and variable wind conditions to verify that no degradation exists.

(6) Compliance with § 27.1121(f) can be accomplished by ensuring that the drain will discharge positively and is a minimum of 0.25 inches in diameter. No drain may discharge where it might cause a fire hazard. This can be demonstrated by discharging a colored liquid through the drain system in flight and on the ground. The dye should not impinge on any ignition source.

(7) Section 27.1121(g) is self-explanatory in specifying that a means must be provided to prevent blockage of the exhaust port after any internal heat exchanger failure. Compliance can be shown by demonstration or by analysis. In either case, it must be shown that any internal failure will not result in a significant power loss from the engine.

AC 27.1123. § 27.1123 (Amendment 27-11) EXHAUST PIPING.

a. Explanation. This section contains the following requirements that must be met for proper certification of exhaust piping on engines, auxiliary propulsion units (APU), and other similar devices.

(1) Section 27.1123(a) requires that the piping be heat and corrosion resistant so that it performs its intended function during its operational life (either the life of the rotorcraft or a specified limited life) without significant metal corrosion, metal erosion, or creation of hazardous hot spots. The piping system should be designed, have an installation design, or a combination that allows performance of its function without thermal expansion (thermal strain) induced structural failures such as ruptures caused by operating temperature excursions and overpressurization during its operational life.

(2) Section 27.1123(b) requires that the piping be supported to withstand the vibration and loading environment (including inertia loads) to which it will be subjected in service.

(3) Section 27.1123(c) requires that piping that connects to components between which relative motion exists in service must have the necessary flexibility and structural integrity to withstand the relative motion without exceeding limit load (at the maximum operating temperature) of the piping, or creating unintended loads (or load paths) on the components to which the piping connects.

b. Procedures. Exhaust piping is typically certified by analysis and installation tests conducted during the basic certification process, including flight tests, as follows:

(1) For compliance with § 27.1123(a), because of its durability in the hot exhaust environment, exhaust piping is typically made from stainless steel or alloy steel of the appropriate structurally and thermally derived wall thickness. Hot aircraft exhaust gases are very corrosive; thus, proper material selection and corrosion protective

design should be performed and validated during certification. Advisory Circular (AC) 43-4, “Corrosion Control for Aircraft” contains a detailed discussion of exhaust gas corrosion problems. Analysis and/or verification tests of the exhaust system should be conducted. This work is necessary to ensure thermal and structural integrity; to ensure that thermal expansion does not cause a structural overload or failure; and, to ensure that exhaust piping does not contact (or come close to) ambient temperature materials (such as structure or system components). Hot exhaust piping in contact with (or close to) ambient temperature materials can either create a fire hazard or cause an unintended strength reduction. To ensure that thermal expansion analyses and tests are properly conducted, the maximum in-service temperature excursion should be properly defined. The maximum temperature excursion should be based on the maximum temperatures of the piping and exhaust gases, as affected by the insulatory characteristics of the piping’s enclosure, and as affected by a worst case hot day. The worst case temperature environment used for analysis can be verified by a temperature survey. If run on cooler days, the survey can be adjusted for the worst case hot day environment using methods identical to those used for engine cooling tests (reference paragraph AC 27.1043, Cooling Tests). The piping should be designed to expand freely so that thermal expansion (thermal strain) induced loads on the piping and its restraint system are minimized. If thermal expansion induced loads (in conjunction with deflection induced loads and exhaust flow loads, discussed in b(4)) are significant relative to the limit load of any item in the load path, then a fatigue check on the critical design point(s) should be performed. The fatigue check should establish a safe life or an approved limited life for the critical component(s) in the system. An accurate analytical fatigue check on exhaust piping may be difficult to perform because of in-service erosion, corrosion, etc.; therefore, phased inspections should be considered to ensure the continued airworthiness of the exhaust piping.

(2) For compliance with § 27.1123(b), exhaust piping should be properly supported so that the maximum loads anticipated in-service are properly distributed and reacted, and as previously discussed, so that thermal expansion induced loading is minimized. Typically the worst case static design load conditions are either the inertia loads from an emergency impact (reference § 27.561) or the combined loading from thermal expansion, in-flight deflections and internal exhaust gas flow (see paragraph b(4)). It should be noted that several combinations of these loads should be examined to determine the critical combination. The piping should be supported and restrained such that critical frequencies are avoided and the induced vibration environment’s effect is minimized. Flight test vibration surveys may be necessary, in some cases, to properly define or validate the critical modes and environment and their effect on the exhaust piping design. Operating modes such as ground idle, flight idle, 40 percent and 80 percent of maximum continuous power, maximum continuous power, OEI power settings and other power settings should be investigated to determine their vibratory effect on the exhaust gas piping system. The strength reduction of the piping materials at operating temperature (and at worst case temperature) should be properly considered in the design and structural substantiation. MIL-HDBK-5D contains material allowables versus temperature data for a wide variety of metallic engineering materials.

(3) For compliance with § 27.1123(c), the piping and its restraint system should be designed to minimize loading induced on the piping by the relative motion (in-service deflections) of the components to which the system attaches. Isolation of significant deflection induced loading (if required based on analysis and strain surveys) by use of flexible joints or other equivalent devices or designs should be considered. Any such in-line device used to reduce deflection loading should be fireproof and leak free when performing its intended function.

(4) For critical load case determination, the expansion induced thermal loading should be added in with mechanical relative motion induced loads and internal exhaust gas flow loads to provide total critical load for both a proper static and a proper fatigue structural substantiation. The critical combined static load should be compared with the emergency impact loads of § 29.561(paragraph b(2)) to determine the critical design load case for static strength substantiation.

(5) It should be noted that the majority of the exhaust piping verification testing required for certification can be accomplished during the rotor drive system tie down testing of § 27.923.

SUBPART E - POWERPLANT

POWERPLANT CONTROLS AND ACCESSORIES

AC 27.1141. § 27.1141 (Amendment 27-12) POWERPLANT CONTROLS: GENERAL.

a. Explanation.

(1) Section 27.1141(a) references §§ 27.777 and 27.1555. The detailed compliance procedures for powerplant controls arrangement and markings are found in these sections.

(2) Each flexible powerplant control should be approved.

(3) In order to prevent power failure due to improper powerplant control valve positioning, § 27.1141(c) specifies acceptable open/closed positions for manual valves. Power-assisted valves should have means to indicate to the flightcrew that the valve is either in the fully open or fully closed position or that the valve is moving between these two positions.

(4) For turbine installations, no single failure or malfunction, or probable combination thereof, of any powerplant control system should cause the failure of any powerplant function necessary for safety.

b. Procedures.

(1) Procedures for § 27.1141(a) are contained in detail in §§ 27.777 and 27.1555.

(2) Compliance with § 27.1141(b) may be accomplished by qualifying the control to Mil-C-7958, “Controls, Push-Pull, Flexible, and Rigid,” or other approved standards.

(3) Compliance with § 27.1141(c)(1) may be accomplished by installing manual valves which have positive stops in the full open and closed positions. The fuel valves, however, may have an arrangement to facilitate the capability of switching to different fuel tanks if suitable indexing is provided. Compliance with paragraph (c)(2) may be accomplished by installing a device which displays to the flightcrew one indication with valve fully open and another with the valve fully closed. Alternatively, an indication could be given when the valve is moving from fully open to fully closed with the indication ceasing when the valve position corresponds to the selected switch position (open or closed). An example would be a light that is off when the valve is fully open or closed and illuminates while the valve is transitioning.

(4) Compliance with § 27.1141(d) can be accomplished by performing a failure mode and effects analysis (FMEA) to determine that no single failure or malfunction will cause failure of any powerplant control function necessary for safety. Included in this FMEA should be calculations showing the likelihood of any combination of failures of the powerplant control systems that would cause failure of any powerplant function necessary for safety is improbable. One acceptable procedure for documenting the analysis is contained in Society of Automotive Engineers (SAE) Fault/Failure Analysis Procedure ARP 926A, revised November 15, 1979.

AC 27.1141A. <u>§ 27.1141 (Amendment 27-23) POWERPLANT CONTROLS: GENERAL</u>.

a. <u>Explanation</u>. Amendment 27-23 changed § 27.1141(c) to extend its applicability to any powerplant valve regardless of the location of the valve control. The previous rule was only applicable for valves in the cockpit. Valves are excluded if their function is not required for safety.

b. <u>Procedures</u>. This rule change did not change the suggested methods of compliance.

AC 27.1143. <u>§ 27.1143 (Amendment 27-11) ENGINE CONTROLS</u>.

a. <u>Explanation</u>. This regulation describes the arrangement and operation of the engine controls.

(1) Each throttle mechanism should be independent of the throttles for other engines.

(2) The arrangement of the independent throttles should allow simultaneous control of all engines with one hand.

(3) Immediate actuation at the engine control should be provided by any given input at the throttle control in the cockpit.

(4) If throttle controls incorporate a fuel shut-off feature, a means should be provided to prevent inadvertent movement to the shut-off position. This means should--

(i) Provide a positive lock or stop at the idle position. An idle detent (mechanical or electrical/mechanical such as solenoid) is an accepted arrangement.

(ii) Require a separate and distinct operation to place the control in the shut-off position. Separate action (switch or button) to displace the idle stop or distinct offsets in throttle motion to allow movement from the idle stop to shutoff are accepted arrangements.

b. <u>Procedures</u>. None.

AC 27.1143A. <u>§ 27.1143 (Amendment 27-23) ENGINE CONTROLS</u>.

a. <u>Explanation</u>. Amendment 27-23 revises § 27.1143 by replacing the terms "throttle control" and "thrust control" with the more general term "power control." The changes should preclude misconceptions regarding engine control arrangements when governor-controlled turboshaft engines are employed in rotorcraft.

b. <u>Procedures</u>.

(1) Proper operation of the power control functions should be verified as part of the Type Inspection Authorization (TIA).

(2) Compliance with § 27.1143(d)(1) has been shown successfully in the past by using idle detentes (mechanical or electrical/mechanical, such as a solenoid).

(3) Compliance with § 27.1143(d)(2) has been achieved by using a switch or button to displace the idle stop. Distinct offsets in throttle motion to allow movement from the idle stop to shutoff have also been used to show compliance.

AC 27.1143B. <u>§ 27.1143 (Amendment 27-29) ENGINE CONTROLS</u>.

a. <u>Explanation</u>. Amendment 27-29 introduced the option of using 30-second/2-minute OEI power ratings to multiengine rotorcraft. This amendment revises § 27.1143 by adding the requirement for automatic control of 30-second OEI limits in the new § 27.1143(e). Automatic control of the 30-second OEI limits are required to prevent exceeding the remaining power sections OEI limits after the precautionary shutdown of one engine. The use of 30-second OEI power must be limited to emergency use only during flight conditions where one engine has failed or has been shutdown for precautionary reasons. During this critical stage of flight, crew attention should not be focused on powerplant instruments to avoid exceeding the limit.

b. <u>Procedures</u>. The automatic controls used to prevent 30-second OEI limit exceedances can be installed on the airframe or the engine. The applicant should demonstrate that 30-second OEI limits that can affect the continued operation of the drive system or engine such as gas generator speed, power turbine speed, measured gas temperature, torque, etc., cannot be exceeded. It should also be shown that these devices do not restrict the ability to achieve the full 30-second OEI limits. The operation of these limit devices can be demonstrated on the aircraft or if possible by using bench tests.

AC 27.1145. <u>§ 27.1145 (Amendment 27-12) IGNITION SWITCHES</u>.

a. <u>Explanation</u>.

(1) This section addresses the arrangement and protection of ignition switches for reciprocating engines or for turbine engines which require continuous ignition.

(2) The objective is to provide a means to quickly shut off all ignition, if required, while at the same time providing protection against inadvertent ignition switch operation.

(3) Section 27.1145(a) does not specifically state that turbine engines which do not require continuous ignition are excluded from the rule, but no benefit is realized by the capability of shutting off all ignition to these engines.

b. <u>Procedures</u>.

(1) Section 27.1145(a) is self-explanatory in specifying that a means be available to quickly shut off all ignition by the grouping of switches or by a master ignition switch control. A “T” arrangement or split rocker switches are possible configurations. A master ignition control, if utilized, would need to be carefully evaluated if rotorcraft performance credit is given for engine isolation.

(2) Each group of ignition switches and the master ignition control should have a means to prevent inadvertent operation. “Guarded” switches are the usual means of showing compliance.

AC 27.1147. <u>§ 27.1147 MIXTURE CONTROLS</u>.

a. <u>Explanation</u>. This section addresses the arrangement of fuel mixture controls for reciprocating engine installations and applies only if mixture controls are installed. Note that this control, as used in rotorcraft, is an engine shutdown device. Adjustment of the fuel mixture in flight is not allowed to demonstrate Part 27 compliance, but may be acceptable for more efficient engine operation if suitable stops or automatic means are provided to prevent inadvertent engine shutdown with mixture movement or engine malfunction with flight condition changes.

b. <u>Procedures</u>.

(1) The arrangement should allow--

(i) Separate control of each engine; and

(ii) Simultaneous control of all engines.

(2) Compliance may be accomplished by a side-by-side arrangement of the controls to allow either separate or simultaneous control.

AC 27.1151. <u>§ 27.1151 (Amendment 27-33) ROTOR BRAKE CONTROLS</u>.

a. <u>Explanation</u>.

(1) Amendment 27-33 added a new § 27.1151 that establishes requirements for rotor brake controls. Paragraph a is intended to require design features which, for all practicable purposes, prevent inadvertent brake application in flight even under conditions of reasonably expected crew error or confusion.

(2) Paragraph b requires warning devices to alert the crew if the brake has not been completely released.

b. <u>Background</u>. Inadvertent or undetected application of the rotor brake may be expected to result in excessive heat and fire in the rotor brake area. Rotor brake components are usually located integral with, or in close proximity to, rotor drive system components and, in some cases, close to critical hydraulic main rotor control system components. Fires in these areas would be extremely hazardous.

c. <u>Method of Compliance</u>.

(1) For paragraph (a), literal compliance can be achieved by lock-out devices sensitive to the higher RPM range of the main rotor or other flight parameters, hydraulic bypass or lock-out devices controlled by flyweight governor systems, or engines control position, etc.

The guard required by FAR 27.921 does not, in itself, provide compliance with this requirement. However, if careful evaluation of the overall control, including location, guard mechanism, control manipulation requirements, accessibility, etc., provides a high degree of assurance that inadvertent application will not occur, compliance may be assumed. Also, if brake application does occur, annunciation appears and no immediate hazard to flight operation exists, compliance may be assumed.

(2) Alerting devices supplied to comply with this rule should provide a signal at any time the rotor brake is engaged, including partial engagement. This means to alert the crew could be:

- A warning light indicating the mechanical control position, or the position of the brake for a power assisted system, or

- An unambiguous device warning the crew that the rotor brake is engaged (or partially engaged), or

- A locking device preventing the engine starting when the rotor brake control is not completely released.

AC 27.1163. <u>§ 27.1163 (Amendment 27-23) POWERPLANT ACCESSORIES</u>.

a. <u>Explanation</u>.

(1) This section addresses the interface requirements for powerplant accessories which are mounted on the engine or rotor drive system components.

(2) Areas which should be addressed include structural loads imposed upon the engine case and isolation between the accessory and engine oil systems. Electrical equipment isolation from flammable fluids or vapors should be addressed as well as the effect of an accessory failure on the continued operation of the engine and drive system components.

b. <u>Procedures</u>.

(1) Accessories installed and certified by the engine manufacturer can be mounted on the engine without additional justification.

(2) Any accessory to be mounted on the engine, which was not certificated with the engine, and does not meet the engine installation design manual requirements should have a structural analysis showing the mounting of that accessory on the engine will not induce loads into the engine case which are higher than the original design loads.

(3) When the accessory is mounted and operating on the engine, it should not be possible to contaminate either the engine or accessory oil systems. This contamination can take the form of debris following a failure, airborne dirt or water, or any other substance that would impair proper operation of the engine or accessory. Compliance with these requirements can be accomplished by a combination of test and analysis. The design interface should be such that when the equipment is operating, there are no high/low pressure differentials between the components which would induce fluid transfer between components resulting in a low fluid level in one component and an overfill condition in the other component. Where this potential exists, an analysis and/or test should be used to demonstrate compliance.

(4) Engine mounted accessories which are subject to arcing and sparking, must be isolated from all flammable fluids or vapors to minimize the probability of fire. This can be accomplished by isolating the electrical equipment from the flammable fumes or vapors or by isolating the flammable fumes or vapors from the potential ignition source. Compliance can be shown by analysis.

(5) A failure mode and effect analysis should be submitted which shows that a failure of any engine mounted and driven accessory will not interfere with the continued

operation of the engine. If a hazard is created by the continued rotation of an engine driven accessory after a failure or malfunction, provisions to stop its rotation or eliminate the hazard must be provided. The effectiveness of this device should be demonstrated by test.

(6) The main transmission and rotor drive system should be protected from excessive torque loads and damage imposed upon them by accessory drives. One method which has been used is a torque limiting device; (i.e., shear section of main rotor driveshaft). The effectiveness of any protection device should be demonstrated by test.

SUBPART E - POWERPLANT

POWERPLANT FIRE PROTECTION

AC 27.1183. § 27.1183 (Amendment 27-20) LINES, FITTINGS, AND COMPONENTS.

a. Explanation. This section requires that any line, fitting or other component of a flammable fluid, fuel or flammable gas system which carries, conveys, or contains the fluid or gas in any area subject to engine fire conditions (i.e., a severe fire) must be at least fire resistant (reference § 1.1 for definition of fire resistant and see paragraph AC 27.859 which defines a severe fire). An exception is for flammable fluid tanks and supports which are part of and attached to the engine or are in a designated fire zone. These items are required to either be fireproof (see § 1.1 for definition of fireproof and see paragraph AC 27.859 which defines a severe fire) or to be enclosed by a fireproof shield, unless fire damage to any non-fireproof part (e.g., secondary line or valve support) will not cause leakage of a flammable gas, flammable fluid or otherwise prevent continued safe flight and landing of the rotorcraft. All such components must be shielded, located, otherwise protected or a combination to safeguard against the ignition of leaking flammable fluids or gases. Integral oil sumps of less than 25 quarts capacity on a reciprocating engine need not be fireproof or enclosed by a fireproof shield; however, they should be fire resistant. Most integral sumps in this category are, by natural design and material selection, fire resistant. Exemptions to the preceding requirements are as follows:

(1) Lines, fittings and components already approved under Part 33 as part of the engine itself.

(2) Vent and drain lines (and their fittings) whose failure will not result in or add to an operational fire hazard. In addition, all flammable fluid drains and vents must discharge clear of the induction system air inlet and other obvious ignition hazards.

b. Procedures. A detailed review of the design should be conducted to identify and quantify all lines, fittings, and other components which carry flammable fluids and/or gases and are in areas subject to engine fire conditions such as engine compartments and other fire zones. Once these items are identified the design means of fire protection should be selected and validated, as necessary, during certification. For materials and devices that cannot be qualified as fireproof or fire resistant by similarity or by known material standards, testing to severe fire conditions (see paragraph AC 27.859 definition, AC 20-135, and AC 23-2 for detailed requirements) should be conducted on full-scale specimens or representative samples to establish their fireproof or fire resistance capabilities. Exceptions to these standards (as provided in the regulatory section) should be reviewed and approved/disapproved on a case-by-case basis during certification. Also, operational fire hazards from drains, vents, and other similar sources should be identified and eliminated during certification.

AC 27.1185. § 27.1185 (Amendment 27-11) FLAMMABLE FLUIDS.

a. Explanation. This section requires that fuel, flammable fluid, or vapor tanks, reservoirs or collectors be sufficiently isolated from engines, engine compartments, and other designated fire zones so that hazardous heat transfer from these areas to fuel, flammable fluid, and vapor tanks, reservoirs, or collectors is prevented in either normal or emergency service.

b. Definitions.

(1) Fuel or Flammable Fluid Collector. Any device such as a large valve, accumulator, or pump that contains a significant amount of flammable fluid, fuel, or vapor (e.g., the volume equal to 10 ounces or more of fluid).

(2) Flammable Fluid or Vapor Tank. Any fuel, flammable, fluid, or vapor tank, reservoir, or collector.

(3) Sufficiently Isolated. Fuel, flammable fluids, or vapors in a tank, reservoir, or collector are insulated, removed, otherwise protected or a combination such that their worst case temperatures (the worst case measured or calculated surface temperature of their containers) in either normal or emergency service is always 50° F or more away from the autoignition temperature of the fuel, flammable fluid, or vapor in question.

(4) Minimum Autoignition Temperature. The temperature at a given vapor pressure at or above which liquid fuel or fuel vapor will self combust. When determining the minimum design value of autoignition temperature which will occur in either normal or emergency operations, the critical, in-service combination of vapor pressure and fuel temperature should be determined.

(5) Hazardous Heat Transfer. A total incident heat flux (a combination of conduction, convection, and radiation, as applicable) from or in an engine compartment or other designated fire zone, which would raise the temperature level of a flammable fluid or fuel, their vapors, or the surface temperature of their containers to within 50° F or less of the minimum in-service autoignition temperature. Typically, the most critical heat transfer case to be considered is emergency service where a severe fire (see definition) is assumed to occur in each engine compartment and each designated fire zone on a case-by-case basis.

(6) Severe Fire. See definition in paragraph AC 27.859.

c. Procedures.

(1) The fuel, flammable fluid, and vapor system designs should be reviewed early in certification to insure that all flammable fluid or vapor tanks are properly identified and isolated from engines, engine compartments, and other designated fire

zones during both normal and emergency operations such as in-flight engine compartment or other fire zone fires. In some cases fuel or flammable fluid components must be located in an engine compartment or other designated fire zone. In these cases, an equivalent safety finding (which considers the design, construction, materials, fuel lines, fittings, and controls used in the system, or system segment, contained in the engine compartment or other designated fire zone) should be undertaken as a part of the normal certification process. If the level of safety provided is equivalent to that provided by removing the system or system segment out of the engine compartment or designated fire zone, then the design should be accepted. For fuel tanks only, isolation is required by regulation to be achieved by use of either a firewall (reference paragraph AC 27.1191 for Firewall Requirements) or by use of a shroud. A shroud if used should be fireproof (see § 1.1 for definition and the definition of a Severe Fire for further details) and should be drainable (or otherwise inspectable) to insure the fuel tank is not leaking in service. For other flammable fluid or vapor tanks, the regulations allow either the identical treatment previously described for fuel tanks (i.e., firewalls or shrouds) or, alternatively, use of an equivalent safety finding. The equivalent safety finding, if used, can be made as part of the standard certification process. Regulations require that the equivalent safety finding be based on system design, tank materials, tank supports, and flammable fluid system connectors, lines, and controls. In all cases the flammable fluids, fuels, and vapors should be sufficiently isolated from hazardous heat fluxes during both normal and emergency operations to prevent autoignition.

(2) In addition, the regulations require at least one-half inch of clear airspace between each flammable fluid or vapor tank and each firewall or shroud that isolates the system, unless equivalent means (such as fireproof insulation) are used to prevent hazardous heat transfer from each engine compartment or other fire zone to the flammable fluid or vapor mass (or its container surface) at the fluid or vapor’s minimum autoignition temperature. If in-service structural deflections are significant, they must be taken into account when certifying the one-half inch minimum clear airspace requirement. For example, if a one-half inch clearance exists on the ground but in some normal and emergency flight conditions (e.g., autorotation) the one-half inch is reduced to one-fourth inch at a critical time (in-flight engine fire), then the design (static) configuration should have at least a one-half plus one-fourth equals three-fourths inch static clear airspace to insure the regulation’s intent is met. Alternatively, fireproof insulation or additional stiffeners could be used to insure the regulation’s intent is met (i.e., the thermal equivalent of one-half inch clearance is maintained at all times). Any material used as insulation on or used adjacent to a flammable fluid or vapor tank, should be certified as chemically compatible with the flammable fluid or vapor and to be non-absorbent in case of fuel or vapor leaks. Otherwise, the material should either be treated for compatibility and non-absorbency or not accepted.

AC 27.1187. <u>§ 27.1187 VENTILATION</u>.

a. <u>Explanation</u>. To ensure that any component malfunction which results in fuel, flammable fluid or vapor leaks is safely drained or vented overboard and to ensure that a fire hazard is not created during either normal or emergency service, there should be complete, rapid drainage and ventilation capability present for each part of the rotorcraft powerplant installation and any other designated fire zone which utilizes flammable fluid or vapor carrying components. As a minimum, the routing, drainage, and ventilation system should accomplish the following:

(1) It should be effective under normal and emergency operating conditions.

(2) It should be designed and arranged so that no discharged fluid or vapor will create a fire hazard under normal and emergency operating conditions.

(3) It should prevent accumulation of hazardous fluids and vapors in engine compartments and other designated fire zones.

b. Definitions. Drip Fence. A physical barrier that interrupts the flow of a liquid on the underside of a surface, such as a fuel tank, and allows any leaked liquid to drip from the surface away from hazardous locations to a safe external drain.

c. Procedures. The design of flammable fluid and gas systems running through engine compartments and other designated fire zones should have a thorough hazard analysis performed early during certification that is updated periodically as design changes dictate. The hazard analysis should identify and quantify all normal and emergency service failures that could result in leakage of fuel, flammable fluids and vapors. Once these potential hazards are identified and quantified, appropriate design features, such as drains, drip fences and vents, that minimize or eliminate the hazard should be provided. These means should be analyzed, tested, or a combination as necessary, to ensure that their size, flow capacity, and other design parameters are adequate to rapidly remove hazardous fluids and vapors safely away from the rotorcraft under normal and emergency flight conditions. Typically a venting or draining system should be designed to a 3-to-1 flow capacity margin over the probable worst case leak to which it could be subjected. Adverse effects such as clogging and surface tension flow reduction should be accounted for in design. Testing, including flight testing, using inert fluids or vapors may be necessary for proper design certification. In some instances it may be appropriate to include ventilation and drainage tests when the aircraft is parked.

AC 27.1189. § 27.1189 (Amendment 27-23) SHUTOFF MEANS.

a. Explanation.

(1) This section establishes the requirements for controlling hazardous quantities of flammable fluids which flow into, within, or through designated fire zones.

(2) When any shutoff valve is operated, any equipment, including a remaining engine, which is essential for continued flight, cannot be affected.

b. Procedures.

(1) Combustible fluid supply lines which pass into, within, or through a firewall into the fire zone must incorporate shutoff valves. This requirement does not apply to lines, fittings, and components which were certified with and are part of the engine. These requirements do not apply to oil systems for reciprocating engines with less than 500 cubic inches displacement or to any other installation where all components, including the oil tanks, are fireproof or are located in an area that will not be affected by an engine fire.

(2) Eight fluid ounces or less of a combustible fluid is not considered hazardous and no more than this amount should be present after activating the shutoff valve.

(3) Engine isolation is to be maintained when incorporating shutoff valves into engine fuel and lubrication lines. The design should ensure that when one engine is shut down or fails and the fuel and lubrication fluid shutoff valves are activated, the remaining good engine is not affected in any way, and the rotorcraft can continue safe flight to a landing. This should be demonstrated by test.

(4) Each shutoff valve located in a fire zone should be fireproof. If the shutoff valve is located outside of the fire zone, then it should be at least fire resistant or protected so that it will function under a worst case fire condition within a fire zone. This should be demonstrated by test.

(5) For primary propulsion engine installations, the flammable fluid shutoff should be protected from inadvertent operation. Where electrical shutoffs are used, the switches should be guarded or require double actions. If the shutoffs are mechanically activated, the design of the knob and the location of the lever should be such that inadvertent actuation cannot occur. It must be possible to reopen the shutoff valve after it has been closed and this should be demonstrated by test.

AC 27.1191. § 27.1191 (Amendment 27-2) FIREWALLS.

a. Explanation. This section states the certification requirements for the use of fireproof protective devices such as firewalls, shrouds, or equivalent. These devices are necessary to isolate each engine (including combustor, turbine, and tailpipe sections of turbine engines and auxiliary propulsion units (APU); each APU; each combustion heater; each unit of combustion equipment; or each high temperature device (or source) from personnel compartments and critical components (not already protected under § 27.861). The isolation of these fire zones is necessary to prevent the spread of fires, prevent or minimize thermal injuries and fatalities, and prevent damage to critical components that are essential to a controlled landing. Even though § 27.1191(b) implicitly excludes APU’s, combustion heaters, and other combustion equipment that are not used in flight; they should be protected by fireproof enclosures, because of the requirements of the relevant parts of §§ 27.1183 through 27.1203. This is because, even if the device is rendered inoperative in flight, it typically contains residual heat, fuel, fumes and potential ignition sources (i.e., "potential hazards"). Each fireproof protective device must, by regulation, meet the following criteria:

(1) Its design and location must take into account the probable fire path from each fire zone or source considering factors such as internal airflow, external airflow, and gravity.

(2) It must be constructed so that no hazardous quantity of air, fumes, fluids, or flame can propagate through it to unprotected parts of the rotorcraft.

(3) Its openings (e.g., shaftholes, lineholes, etc.) must be sealed with close fitting fireproof grommets, bushings, bearings, firewall, fittings, or equivalent that prevent burn through and leakage of hazardous fumes or fluids from the fire zone.

(4) It must be fireproof (see definition).

(5) It must be either corrosion resistant or otherwise safely protected from corrosion.

b. Definitions.

(1) Fireproof Protective Device. A fireproof protective device is a device such as a firewall, shroud, enclosure, or equivalent used to isolate a heat or potential fire source (severe fire) from personnel compartments and from critical aircraft components which are essential for a controlled landing.

(2) Fireproof. Fireproof is defined in § 1.1 "General Definitions."

(3) Controlled Landing. A landing which is survivable (i.e., does not fatally injure all occupants) but may produce an unairworthy, partially salvageable, or unsalvageable rotorcraft.

(4) Severe Fire. See Definition in paragraph AC 27.859.

c. Procedures. Fireproof protective devices are typically certified by analysis, tests, or a combination conducted during the certification process, including flight tests or simulated flight tests, as follows:

(1) Fireproof protective devices should be provided wherever a hazard exists which requires isolation from a severe fire to avoid fires in personnel compartments and to avoid thermal damage to critical components (such as structural elements, controls, rotor mechanisms, and system components) that are necessary for a controlled landing. A thorough hazard analysis should be conducted during certification to identify, define and quantify in order of severity (i.e., maximum temperature, hot exposed area, etc.) all thermal hazards or zones that require fireproof protection in a given design. Engines (including the combustor, turbine, and tailpipe sections of turbine engines), APU's,

combustion heaters, and combustion devices are required by regulation to be isolated. Other high temperature devices may also require isolation because of local hot spots (which occur during normal operations or from failure modes) that can thermally injure occupants or cause spontaneous combustion of surroundings. A hazard analysis should identify these potential problems and provide proper certification solutions.

(2) Fireproof protective devices should be able to withstand at least 2000 ± 150° F for at least 15 minutes (reference AC 20-135). The fireproof protective device should allow protected parts, subsystems or systems to perform their intended function for the duration of a severe fire (see definitions). For firewalls, examples of flat, geometry materials undergoing uniform heat fluxes with material gauges that automatically meet the certification requirements are given in figure AC 27.1191–1. If firewalls are utilized that involve other materials, significant geometric changes, or significantly non-uniform heat fluxes, then automatic compliance may not be assured. In such cases the fireproof protective device should be analyzed using the severe fire definition and, in some cases, tested in accordance with AC 23–2 to ensure proper certification. For example, a curved protective surface may absorb a uniform incident heat flux unevenly and create a local hot spot that exceeds 2,150° degrees Fahrenheit that burns through in less than 15 minutes; whereas, a flat surface of equal thickness might not exceed 2,150°degrees Fahrenheit and would not burn through in less than 15 minutes. It should be noted that composite materials are not generally used for protective devices because of their inability to withstand high temperatures (i.e., exceedance of the glass transition temperature); however, some specially formulated composites have been previously certified as engine cowlings. Titanium is an acceptable material for fireproof protective devices such as firewalls. However, use of titanium should always be carefully considered and reviewed, because it can lose all structural ability and burn severely (self combust) above 1,050° F, under certain thermodynamic environments, and contribute to the fire instead of providing the intended fire protection. AC 33–4, “Design Considerations Concerning the Use of Titanium in Aircraft Turbine Engines” and MIL-HDBK-5D contain more detailed information on the unique thermal properties of titanium.

FIGURE AC 27.1191-1
TABLE OF MATERIALS AND GAGES ACCEPTABLE
FOR FIREPROOF PROTECTIVE DEVICES WITH FLAT
SURFACE GEOMETRIES[1]

MATERIAL[2]	MINIMUM THICKNESS[3]
Titanium Sheet	.016 in
Stainless Steel	.015 in
Mild Carbon Steel	.018 in
Terne Plate	.018 in
Monel Metal	.018 in
Firewall Fittings (Steel or Copper Base)	.018 in[4]

NOTES:

(1) Assumes essentially flat vertical or horizontal surfaces undergoing a uniform heat flux. Any significant variation in either geometry or heat flux distribution should be examined in detail for adequate gauge thicknesses on a case-by-case basis.

(2) Must have corrosion protection if not inherent in the material itself.

(3) The minimum thickness is for thermal containment only. Structural integrity considerations may require thickness increases. MIL-HDBK-5D contains material allowable versus temperature data for most common metallic materials.

(4) This is the minimum wall thickness measured at the smallest dimension (e.g., thread root or other location) of the part.

(5) Distortion of thin sheet materials and the subsequent gapping at lap joints or between rivets is difficult to predict; therefore, testing of the simulated installation is necessary to prove the integrity of the design. However, rivet pitches of 2 inches or less on non load-carrying titanium firewalls of .020 inch or steel firewalls of .018 inch are acceptable without further testing.

(3) The probable path of a fire (as affected by internal and external air flow during normal flight and autorotation, gravity, flame propagation paths, or other considerations) should be taken into account when performing the hazard analysis of item (1). Such a review will ensure that fireproof protective devices are placed in the proper location for intercepting, blocking or containing a severe fire before occupants are injured and a controlled landing is prevented. If the probable path cannot be readily determined by inspection or analysis, testing using simulated air flows, rotorcraft attitudes, and dyed inert fluids or vapors can be used to aid in this determination.

(4) Each opening in a protective device should be sealed with close fitting sealing devices such as fireproof grommets, bushings, firewall fittings, rotating seals or equivalent that are at least as effective as the fireproof protective device itself. This is necessary to ensure that no local breakdowns in protection occur. For materials not listed as acceptable in item (1), analysis and testing should be required in accordance with FAA/AUTHORITY standards and the definition of a severe fire for proper substantiation.

(5) Each protective device should be fireproof in order to withstand a severe fire. Unless designs and materials have been previously FAA/AUTHORITY approved (e.g., see Item 1), the protective device's design and material selection should be tested to ensure its fireproof thermal and structural integrity. A full-scale test of a structurally loaded article or a representative sample should be conducted to ensure proper compliance is achieved. Also, the continued sealing ability of the protective device in its deformed state due to a hard controlled landing should be considered during certification (e.g., use of ductile materials). The corrosion environment should be defined and appropriate protection provided. Phased inspections should be specified, if necessary, to ensure continued corrosion integrity. Certification tests for adequacy of corrosion protection should be conducted using sample plates or by other equivalent means, as required.

AC 27.1193. § 27.1193 COWLING AND ENGINE COMPARTMENT COVERING.

a. Explanation.

(1) Section 27.1193(a) requires the cowling and engine compartment coverings to structurally withstand loads experienced in flight.

(2) In order to prevent pooling of flammable fluids, § 27.1193(b) requires rapid and complete drainage from the cowling and engine compartment.

(3) Section 27.1193(c) requires the drain of paragraph (b) to purge the fluid in such a manner not to create a fire hazard.

(4) Section 27.1193(d) requires the cowling and engine compartment covering to be at least fire resistant and paragraph (e) requires them to be fireproof where they may experience high temperatures due to the exhaust system.

b. Procedures.

(1) Compliance with § 27.1193(a) can be shown by analyzing the cowling and engine compartment covering and determining that no structural degradation will occur under the highest loads experienced on the ground or in flight.

(2) Compliance with § 27.1193(b) can be accomplished by ensuring that the drain will discharge positively with no traps and is a minimum of 0.25 inches in diameter.

(3) Compliance with § 27.1193(c) can be demonstrated by colored liquid flowing through the drain system while in flight. The dye should not impinge on any ignition source during any approved flight regime.

(4) Compliance with § 27.1193(d) can be accomplished by showing that the cowling and engine compartment covering is fire resistant. Fire resistant in this context means a material that has the capacity, under expected service conditions (load, vibration, airflow), to withstand the heat associated with fire at least as well as aluminum alloy in dimensions appropriate for the purpose.

(5) Compliance with § 27.1193(e) can be accomplished by showing that the cowling and engine compartment coverings retain adequate structural integrity when subjected to elevated temperatures that may be expected in service.

AC 27.1193A. § 27.1193 (Amendment 27-23) COWLING AND ENGINE COMPARTMENT COVERING.

a. Explanation. Amendment 27-23 adds a new § 27.1193(f) that requires redundant retention means for each panel, cowling, engine, or rotor drive system covering that can be opened or readily removed. Conventional fasteners for these devices are subject to frequent operation by maintenance personnel and have deteriorated, failed from wear or vibration, or been left unsecured after preflight inspections. Such a failure could be hazardous if a loose panel, cowling, or covering comes in contact with the rotors or critical controls.

b. Procedures. All of the policy material pertaining to this section remains in effect with the following additions:

(1) Compliance with § 27.1193(f) can be accomplished by simulating, or actually failing, one or more of the retention devices or by structural analysis. It should be shown that the cowling or cover will not open, strike, or be struck by the rotor or other critical component.

(2) Consideration should be given to minimize the possibility of latches being improperly closed that could result in a cowl coming open in flight.

(3) The failure of one latching device should not cause the failure of another latching device. If a failure of a single retention device can contribute to multiple failures, these multiple failures should be considered.

(4) The consequences of “forgetting” to latch a cowl should be considered.

(5) The use of safety straps should be considered to minimize the impact of a latching device failure.

AC 27.1194. § 27.1194 (Amendment 27-2) OTHER SURFACES.

a. Explanation. This section states the fire resistance requirements for material surfaces near engine compartments and designated fire zones (other than tail surfaces not subject to heat, flames or sparks emanating from a designated fire zone or engine compartment).

b. Definition.

(1) Other Surface. Any airframe, system, or powerplant component aft of and near an engine compartment, a designated fire zone, or another heat source which would receive a heat flux as a result of a fire in the engine compartment or fire zone that would require the component to be fire resistant.

(2) Fire Resistant. In accordance with § 1.1, is defined as follows:

(i) Sheet metal or structural members with the capacity to withstand the heat associated with the fire at least as well as aluminum alloy in dimensions appropriate for the purpose for which they are used.

(ii) Fluid carrying lines, fluid system parts, wiring, air ducts, fittings and powerplant controls with the capacity to perform their intended functions under the heat and other conditions resulting from a fire.

(3) Fire. A fire in either an engine compartment or a designated fire zone is assumed to occur that produces a heat flux on a system, airframe or powerplant component aft of or near the fire. The effect of each such fire on other surfaces must be considered on a case-by-case basis to determine the critical case. Unless a more rationale definition is furnished and approved during certification, the fire in any engine compartment or designated fire zone should be assumed, for purposes of analysis, to be a severe fire (see definition in paragraph AC 27.859).

c. Procedures.

(1) Other surfaces should be identified during certification by a design review and by a conservative, thorough hazard analysis based on an analytical estimate of the total heat flux (i.e., conduction, convection, and radiation in combination, as applicable)

using the definition of a severe fire and of the resultant “other surface” temperature based on a single fire occurring in each engine compartment and designated fire zone, on a case-by-case basis. Once the other surfaces are identified and their severe fire induced maximum temperatures determined, their configuration and material selection should be reviewed on a case-by-case basis to determine either that they are fire resistant, that they can be made fire resistant (within the limits of practicability), or that it is impracticable to make them fire resistant. If the non-fire resistant other surfaces can be readily made fire resistant they should be. If it is impracticable to make them fire resistant, then they should be relocated, insulated, or a combination in order to reduce the total incident heat flux (and, thus, lower their surface temperature) so that they no longer need be fire resistant. If insulation is used to shield a surface that is subjected to a significant temperature, it must be fire resistant.

(2) A partial validation of analytical heat flux models using the definition of a severe fire can sometimes be achieved during certification tests by using thermocouples or heat-sensitive stickers to measure in-flight temperature ranges and distributions on other surfaces from known thermal environments in engine compartments or other designated fire zones.

AC 27.1195. <u>§ 27.1195 (Amendment 27-5) FIRE DETECTOR SYSTEMS</u>.

a. <u>Explanation</u>.

(1) This section requires quick-acting fire detectors to be installed on turbine powered rotorcraft, when the engine compartment cannot be readily observed in flight by the pilot in the cockpit.

(2) The number of detectors and locations must be sufficient to ensure prompt detection of fire in the engine compartment.

b. <u>Procedures</u>.

(1) The detector system should be designed for highest reliability to detect a fire and not to give a false alarm. It is desirable that it only responds to a fire and misinterpretation with a lesser hazard should not be possible. Engine overtemperature, harmless exhaust leakage, and bleed air leakage should not be indicated by a fire detector system. A fire detection system should be reserved for a condition requiring immediate measures such as engine shutdown or fire extinguishing. There are three general types of detector-procedure systems that are commonly used:

(i) A manual system utilizes warning lights to alert the pilot who then follows prescribed cockpit procedure as a countermeasure. A manual system is adequate for hazards in which a few seconds are not important.

(ii) There is also a semi-automatic system. Occasionally a rotorcraft becomes so complex that the emergency procedure exceeds reasonable expectations of the pilot. In such cases, psychology should be weighted against complexity, and "panic switches," combining multiple procedure functions, should be provided to simplify the mental demands on the pilot. Speed is gained by such designs for hazards which may need it.

(iii) The detector of an automatic system automatically triggers the appropriate countermeasures and warns the pilot simultaneously. Such a system should be carefully evaluated to assure that the advantages outweigh the disadvantages and potential malfunctions.

(2) Fires, or dangerous fire conditions can be detected by means of various existing techniques. The following is a partial list of available detectors:

(i) Radiation-sensing detectors.

(ii) Rate-of-temperature-rise detectors.

(iii) Overheat detectors.

(iv) Smoke detectors.

(v) CO detectors.

(vi) Combustible mixture detectors.

(vii) Fibre-optic detectors.

(viii) Ultraviolet.

(ix) Observation of crew or passengers.

(3) In many rotorcraft it is desirable to have a detection system which incorporates several of these different types of detectors. Radiation-sensing detectors are most useful where the materials present will burn brightly soon after ignition, such as in the powerplant accessory section. Rate of rise detectors are well-suited to compartments of normally low ambient temperatures and low rates of temperature rise where a fire would produce a high temperature differential and rapid temperature rise. It should be noted that under certain circumstances, where a relatively slow temperature increase occurs over a considerable period of time, a fire can occur without detection by rate of rise detectors. Overheat detectors should be used wherever the hazard is evidenced by temperatures exceeding a predicted, set value. Smoke detectors may be suited to low air flow areas where materials may burn slowly, or smolder. Fibre-optic detectors can be used to visually observe the existence of flame or smoke. The three major detector types used for fast detection of fires are the radiation-sensing, rate-of-rise, and overheat detectors. Radiation-sensing detectors are basically "volume"

type which senses flame within a visible space. Overheat-fire detectors can be obtained in either “continuous” or “unit” type.

(4) The detector system should:

(i) Indicate fire within 15 seconds after ignition, and show in which engine compartment the fire is located.

(ii) Remain on for the duration of the fire.

(iii) Indicate when the fire is out.

(iv) Indicate re-ignition of the fire.

(v) Not by itself precipitate or add to the potential of any other hazards.

(vi) Not cause false warnings under any flight or ground operating condition.

(5) Additional features of the detection system are as follows:

(i) A means should be incorporated so that operation of the system can be tested from the cockpit.

(ii) Detector units should be of rugged construction, to resist maintenance handling, exposure to fuel, oil, dirt, water, cleaning agent, extreme temperatures, vibration, salt air, fungus, and altitude. Also, they should be light in weight, small, and compact, and readily adaptable to desired positions of mounting.

(iii) The detector system should operate on the rotorcraft electric system without inverters. The circuit should require minimum current unless indicating a fire or unless a monitoring system is in use.

(iv) Fixed temperature fire detectors should preferably be set at 100° F (37.7° C) to 150° F (65.6° C) above maximum safe ambient temperature, or higher when in compartments where extremely high rate of rise is normally encountered.

(v) Detector system components located within fire zones should be fireproof.

(vi) Each detector system should actuate a warning device which indicates the location of the fire. If fire warning lights are used, they must be in the pilot’s normal field of view.

(vii) Two or more engines should not be dependent upon any one detector circuit. The installation of common zone detection equipment prevents the detection system from distinguishing between the engine installations, necessitating shutting down more than one engine.

(6) The sensing portion of the fire detection system should not extend outside of the coverage area into another fire zone. Detectors, with the exception of radiation-sensing detectors, should be located at points where the ventilation air leaves compartments. If a reverse-flow cooling system is used, detectors should be installed at locations which are outlets under both flight and ground operating conditions. Stagnant air spaces should be avoided and the number of ventilation air exits should be kept to a minimum. Compliance with these recommendations allow the effective placement of a minimum amount of detectors, and still ensure prompt detection of fire in those zones. Radiation-sensing detectors should be located such that any flame within the compartment is immediately sensed. This may or may not be where the ventilation air leaves the compartment.

(7) Fire detectors should be installed in designated fire zones, the combustor, turbine, and tailpipe sections of turbine installations.

(i) Engine Power Section (Combustor, Turbine, and Tailpipe): This zone is usually characterized by predictable hazard areas which facilitate proper detector location. It is recommended that coverage be provided for any ventilating air outlet as well as intermediate stations where leaking combustibles may be expected.

(ii) Compressor Compartment: This is usually a zone of relatively low air flow velocities, but wide geographical possibility for fires. When fire detectors other than radiation-sensing detectors are used, detection at air outlets provides the best protection, and intermediate detector locations are of value only when specific hazards are anticipated.

(iii) Accessory Bullet Nose: Where such a compartment is so equipped that it is a possible fire zone, its narrow confines permit sufficient coverage with one or more detectors at the outlets.

(iv) Heater Detector Location: An overheat detector should be placed in the hot air duct downstream of the heater. If the heater fuel system or exhaust system configuration is such that it is a fire hazard, the compartment surrounding the heater should also be examined as a possible fire zone.

(v) Auxiliary Power Unit Detector Location: The use of a combustion-driven auxiliary power unit creates another set of typical engine compartments defined and treated as above. Some units are so shrouded with fireproof material that these compartments exist only within the confines of the shroud. They are still, however, fire zones and should have a detection system.